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Systems Design Study of the Pioneer Venus Spacecraft

Final Study Report

Appendices to Volume I Sections 3-6 (Part 1 of 3)

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AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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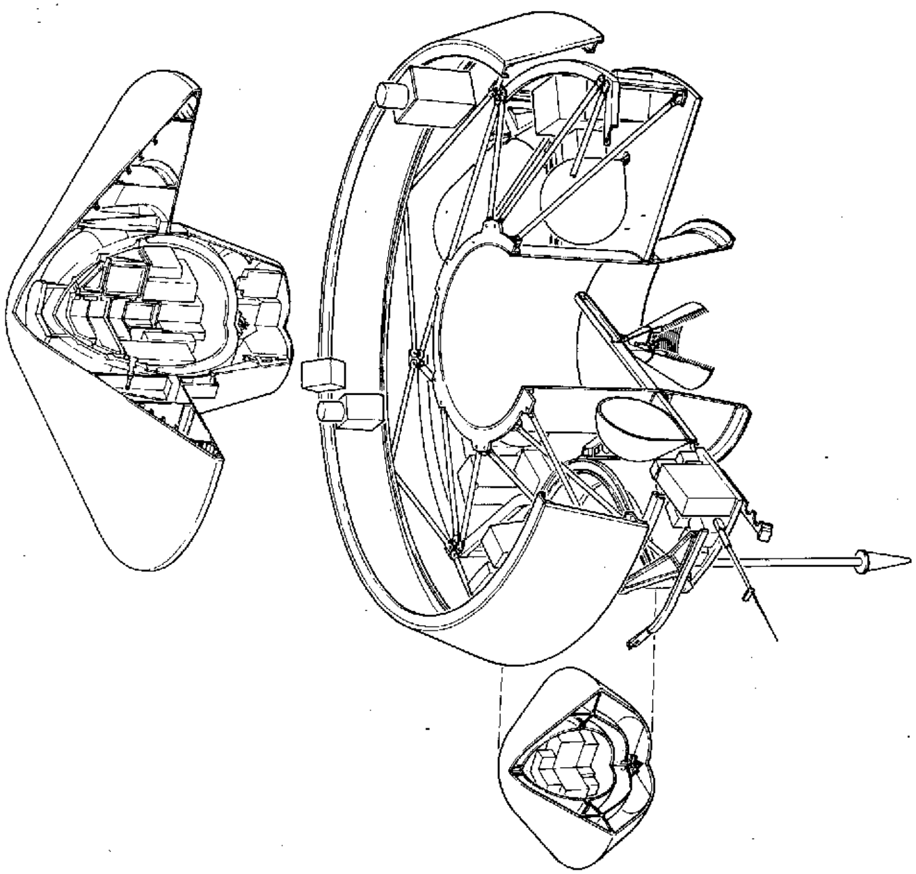

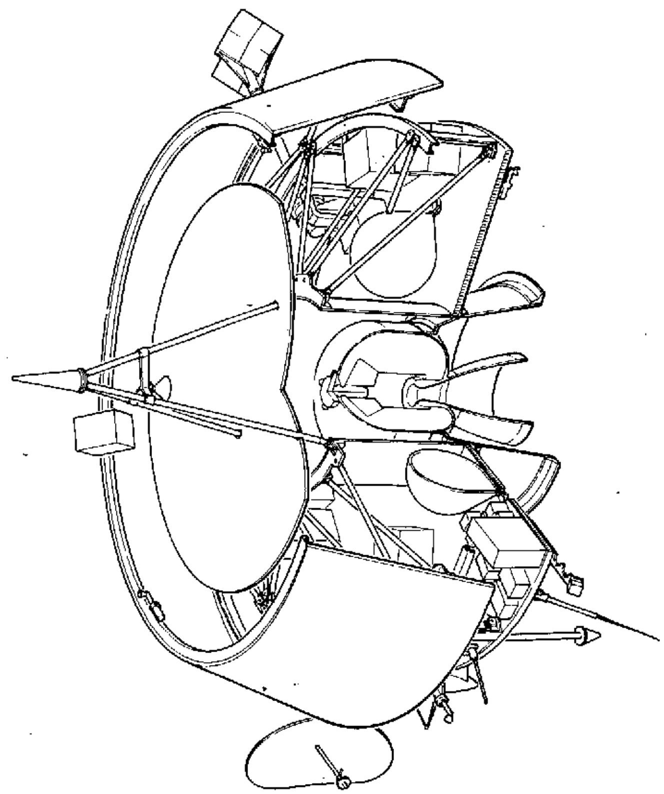

LIST OF VOLUMES

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MARTIN MARIETTA

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ACRONYMS AND ABBREVIATIONS

A	ampere analog
abA	abampere
AC	alternating current
A/C	Atlas/Centaur
ADA	avalanche diode amplifier
ADCS	attitude determination and control subsystem
ADPE	automatic data processing equipment
AEHS	advanced entry heating simulator
AEO	aureole/extinction detector
AEDC	Arnold Engineering Development Corporation
AF	audio frequency
AGC	automatic gain control
AgCd	silver-cadmium
AgO	silver oxide
AgZn	silver zinc
ALU	authorized limited usage
AM	amplitude modulation
a.m.	ante meridian
AMP	amplifier
APM	assistant project manager
ARC	Ames Research Center
ARO	after receipt of order
ASK	amplitude shift key
at. wt	atomic weight
ATM	atmosphere
ATRS	attenuated total refractance spectrometer
AU	astronomical unit
AWG	American wire gauge
AWGN	additive white gaussian noise
B	bilevel
B	bus (probe bus)
BED	bus entry degradation

ACRONYMS AND ABBREVIATIONS (CONTINUED)

BER	bit error rate
BLIMP	boundary layer integral matrix procedure
BPIS	bus-probe interface simulator
BPL	bandpass limiter
BPN	boron potassium nitrate
bps	bits per second
BTU	British thermal unit
C	Canberra tracking station — NASA DSN
CADM	configuration administration and data management
C&CO	calibration and checkout
CCU	central control unit
CDU	command distribution unit
CEA	control electronics assembly
CFA	crossed field amplifier
cg	centigram
c.g.	center of gravity
CIA	counting/integration assembly
CKAFS	Cape Kennedy Air Force Station
cm	centimeter
c.m.	center of mass
C/M	current monitor
CMD	command
CMO	configuration management office
C-MOS	complementary metal oxide silicon
CMS	configuration management system
const	constant construction
COSMOS	complementary metal oxide silicon
c.p.	center of pressure
CPSA	cloud particle size analyzer
CPSS	cloud particle size spectrometer

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ACRONYMS AND ABBREVIATIONS (CONTINUED)

G	Goldstone Tracking Station - NASA DSN gravitational acceleration
g	gravity
G&A	general and administrative
GCC	ground control console
GFE	government furnished equipment
GHE	ground handling equipment
GMT	Greenwich mean time
GSE	ground support equipment
GSFC	Goddard Space Flight Center
H	Haystack Tracking Station - NASA DSN
HFFB	Ames Hypersonic Free Flight Ballistic Range
HPBW	half-power beamwidth
htr	heater
HTT	heat transfer tunnel
I	current
IA	inverter assembly
IC	integrated circuit
ICD	interface control document
IEEE	Institute of Electrical and Electronics Engineering
IFC	interface control document
IFJ	in-flight jumper
IMP	interplanetary monitoring platform
I/O	input/output
IOP	input/output processor
IR	infrared
IRAD	independent research and development
IRIS	infrared interferometer spectrometer
IST	integrated system test
I&T	integration and test
I-V	current-voltage

JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
L	launch
LD/AD	launch date/arrival date
LP	large probe
LPM	lines per minute
LPTTL MSI	low power transistor-transistor logic medium scale integration
LRC	Langley Research Center
M	Madrid tracking station - NASA DSN
MAG	magnetometer
max	maximum
MEOP	maximum expected operating pressure
MFSK	M'ary frequency shift keying
MGSE	mechanical ground support equipment
MH	mechanical handling
MIC	microwave integrated circuit
min	minimum
MJS	Mariner Jupiter-Saturn
MMBPS	multimission bipropellant propulsion subsystem
MMC	Martin Marietta Corporation
MN	Mach number
mod	modulation
MOI	moment of inertia
MOS LSI	metal over silicone large scale integration
MP	maximum power
MSFC	Marshall Space Flight Center
MPSK	M'ary phase shift keying
MSI	medium scale integration
MUX	multiplexer
MVM	Mariner Venus-Mars

ACRONYMS AND ABBREVIATIONS (CONTINUED)

NAD	Naval Ammunition Depot, Crane, Indiana
N/A	not available
NiCd	nickel cadmium
NM/IM	neutral mass spectrometer and ion mass spectrometer
NRZ	non-return to zero
NVOP	normal to Venus orbital plane
OEM	other equipment manufacturers
OGO	Orbiting Geophysical Observatory
OIM	orbit insertion motor
P	power
PAM	pulse amplitude modulation
PC	printed circuit
PCM	pulse code modulation
PCM-PSK-PM	pulse code modulation-phase shift keying-phase modulation
PCU	power control unit
PDA	platform drive assembly
PDM	pulse duration modulation
PI	principal investigator proposed instrument
PJU	Pioneer Jupiter-Uranus
PLL	phase-locked loop
PM	phase modulation
p.m.	post meridian
P-MOS	positive channel metal oxide silicon
PMP	parts, materials, processes
PMS	probe mission spacecraft
PMT	photomultiplier tube
PPM	parts per million pulse position modulation
PR	process requirements
PROM	programmable read-only memory
PSE	program storage and execution assembly

ACRONYMS AND ABBREVIATIONS (CONTINUED)

PSIA	pounds per square inch absolute
PSK	phase shift key
PSU	Pioneer Saturn-Uranus
PTE	probe test equipment
QOI	quality operation instructions
QTM	qualification test model
RCS	reaction control subsystem
REF	reference
RF	radio frequency
RHCP	right hand circularly polarized
RHS	reflecting heat shield
RMP-B	Reentry Measurements Program, Phase B
RMS	root mean square
RMU	remote multiplexer unit
ROM	read only memory rough order of magnitude
RSS	root sum square
RT	retargeting
RTU	remote terminal unit
S	separation
SBASI	single bridgewire Apollo standard initiator
SCP	stored command programmer
SCR	silicon controlled rectifier
SCT	spin control thrusters
SEA	shunt electronics assembly
SFOF	Space Flight Operations Facility
SGLS	space ground link subsystem
SHIV	shock induced vorticity
SLR	shock layer radiometer
SLRC	shock layer radiometer calibration

ACRONYMS AND ABBREVIATIONS (CONTINUED)

SMAA	semimajor axis
SMIA	semiminor axis
SNR	signal to noise ratio
SP	small probe
SPC	sensor and power control
SPSG	spin sector generator
SR	shunt radiator
SRM	solid rocket motor
SSG	Science Steering Group
SSI	small scale integration
STM	structural test model
STM/TTM	structural test model/thermal test model
STS	system test set
sync	synchronous
TBD	to be determined
TCC	test conductor's console
T/D	Thor/Delta
TDC	telemetry data console
TEMP	temperature
TS	test set
TTL MSI	transistor-transistor logic medium scale integration
TLM	telemetry
TOF	time of flight
TRF	tuned radio frequency
TTM	thermal test model
T/V	thermo vacuum
TWT	travelling wave tube
TWTA	travelling wave tube amplifier
UHF	ultrahigh frequency
UV	ultraviolet

ACRONYMS AND ABBREVIATIONS (CONTINUED)

VAC	volts alternating current
VCM	vacuum condensable matter
VCO	voltage controlled oscillator
VDC	volts direct current
VLBI	very long baseline interferometry
VOI	Venus orbit insertion
VOP	Venus orbital plane
VSI	Viking standard initiator
VTA	variable time of arrival
XDS	Xerox Data Systems

SECTION 3 APPENDICES

APPENDIX 3A

VENUS PROBE WINDOWS

APPENDIX 3A
VENUS PROBE WINDOWS

More than 50 percent of the experiments recommended by the Science Steering Group (Reference 1) require probe windows to measure characteristics such as solar flux, infrared flux, aureole, and cloud particles. The equipment inside the probe pressure shell will be kept at a temperature below 345^{o}K; but to avoid condensation on the outside window surface, the window should be above ambient temperature (between 200 and 800^{o}K). This is difficult to achieve because the mechanical window support structure must withstand high pressures, 10 MN/m^2 (100 atm), and therefore tends to require thick walls and high thermal conductivity between window and pressure shell. This, in turn, requires a large amount of electrical power to heat the window above ambient temperature.

A battery mass of 1 kg is required to provide 48 watts of electrical power during a 75-minute probe descent. Because several probe windows are required, power consumption per window should be limited to a few watts. Because the weight penalty for window heating is critical, chemical window heaters have been suggested. This method requires less weight per joule; however, reaction control over the wide temperature range appears to be a problem. Interference with atmospheric composition measurements must also be avoided.

Figure 3A-1 shows a mechanical shutter that opens only when a measurement is performed. The shutter could be combined with a window wiping device. During the measurement the window is flushed with dry carbon dioxide that has passed through an absorption filter or a cold trap. We have successfully operated such a shutter (Figure 3A-2) at 773^{o}K; however, the overall approach to window protection is relatively complex and requires considerable development effort.

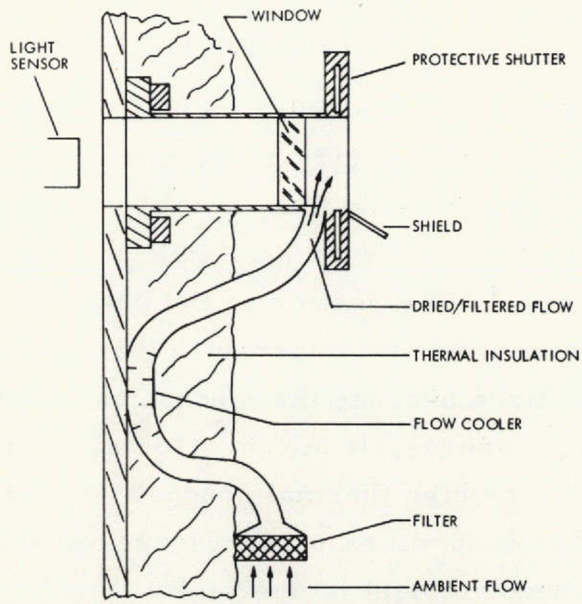

Figure 3A-1. Concept for Window Protection with Shutter and Filtered and/or Cooled Flow

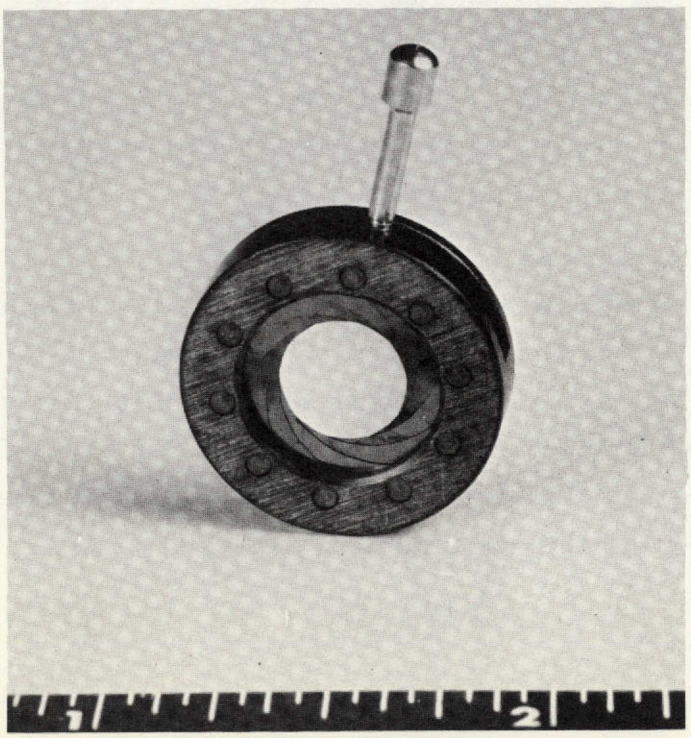

Figure 3A-2. Light Shutter Tested at 755°K

Heated windows with low thermal coupling to the probe pressure shell are considered the most promising approach because of their simple design. The weight penalty for window heating is the most critical tradeoff parameter. Window assembly weight and magnetic interferences are additional considerations. The cavity between the hot outer window and the inner window must be at low pressure ($\leq 10^5$ N/m^2) to avoid considerable convective heat transfer between the windows.

For the window assembly the following design parameters are important.

1) Select a structural material with high strength at 755°K but minimal thermal conduction for the required design parameters. Ductility, very low magnetic permeability, and suitability for bonding to windows are also desirable.

2) Develop a structural design that is compatible with the requirements of high pressure, low weight, and low thermal conduction between window and pressure shell.

3) Select a method of sealing the window(s) to the window assembly and the window assembly to the probe pressure shell. This method must meet the sealing requirements with a minimum weight penalty. A discussion of these design parameters for window assemblies follows.

1. STRUCTURAL MATERIALS FOR VENUS PROBE WINDOW ASSEMBLIES

Kovar can be directly bonded to sapphire but it is magnetic and has low strength at high temperatures. Titanium (TI-6AL-4V) has lower thermal conductivity and thermal expansion characteristics than steel. However, these advantages do not outweigh the superior strength of high temperature alloys such as A-286, Inconel 718, Inconel 750, Flasteloy B, Rene 41 and M252. For example, Inconel 718 has a high yield strength (approximately 9.3×10^8 N/m^2) at 755°K, and its thermal conductivity is similar to those of the other high temperature materials, and it is not magnetic. Therefore, Inconel 718 was selected for metallic support structure of window assemblies.

Glass or ceramics have considerably lower thermal conductivity than the preferred metals. Fused quartz or glass structures for the window assemblies should allow lower thermal conduction than metallic structures. Even if the wall thickness of a quartz tube is twice that of an Inconel 718 tube, the thermal conduction would be less than 30 percent for the quartz tube. We have not yet built a test model with a glass or ceramic support structure because these materials are very brittle. Shock, impact, or thermal distortions of the pressure shell could cause complete failure.

The window material is primarily determined by requirements of transmittance range, structural integrity at high temperatures, compatibility with sealing requirements and chemical resistance to environment. Sapphire was chosen for wavelengths between 0.15 and 5.0 micrometers. The modulus of rupture of sapphire is 4.5×10^8 N/m^2 for up to 1273°K. This material is highly inert to chemical interactions and can be bonded to metals. Temperatures up to 673°K cause only a minor decrease in transmittance at wavelengths longer than 5 micrometers.

For windows with transmittance between 0.5 and 14 micrometers, IRTRAN 2 is a promising candidate. Only about 10 percent degradation has been reported in Reference 2 after exposure to 1028°K for 45 minutes. We have successfully tested an IRTRAN 2 window (Figure 3A-3) with a thickness of 5.74 mm and an aperture of 12.2 mm at 9.3 MN/m^2 and 728°K. No significant change in appearance of the window was observed. If transmittance for longer wavelength is required, IRTRAN 4, IRTRAN 6, and diamond should be considered. The Eastman Kodak Company reports that all IRTRAN materials are generally usable at temperatures up to at least 573°K. Use at higher temperatures is possible depending on conditions of use, type of IRTRAN material, type of atmosphere, duration, pressure, tolerable emissivity, etc.

2. STRUCTURAL DESIGN

Early attempts to bond a sapphire window to a metal flange (Figures 3A-4 and 3A-5) showed that minor distortions of the window flange during mounting caused cracking of the sapphire window. During the search for an improved design of the window assembly, it became apparent that a tubular support structure for the window would not only reduce stresses caused by flange distortions but might provide sufficiently low thermal conduction between the pressure shell and window to make electrical window heating practical.

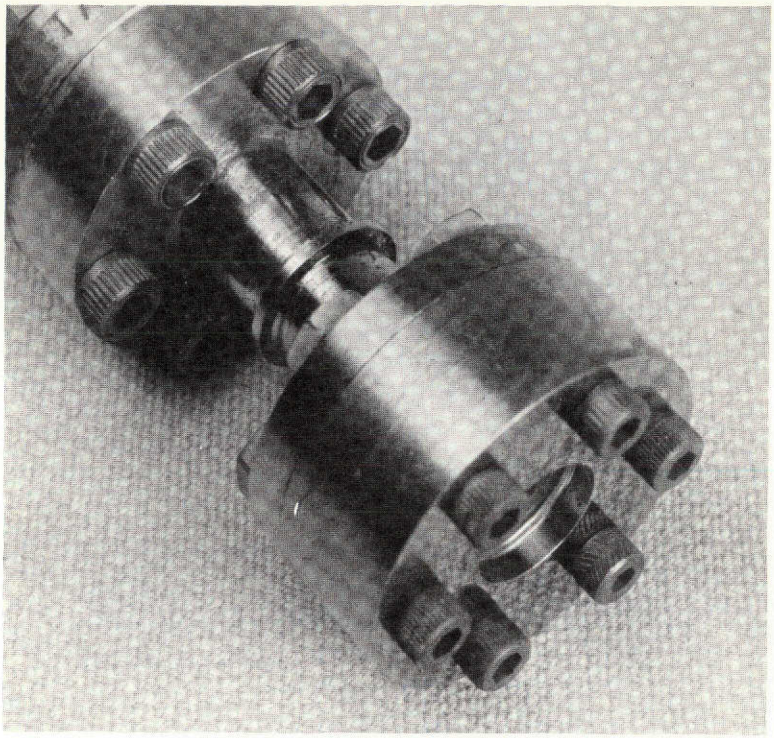

Figure 3A-3. IRTRAN 2 Window Tested at 9.3 MN/m^2 and 728°K

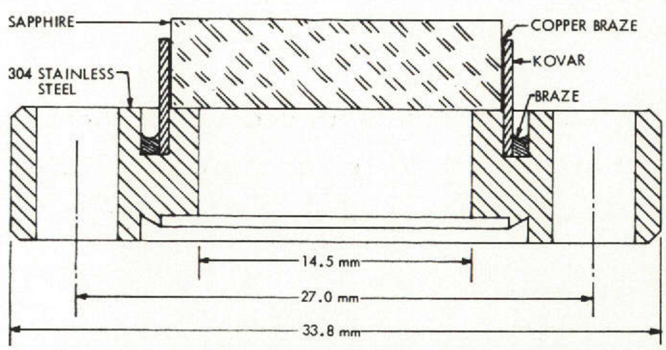

Figure 3A-4. Window Assembly with Kovar to Sapphire Bonding

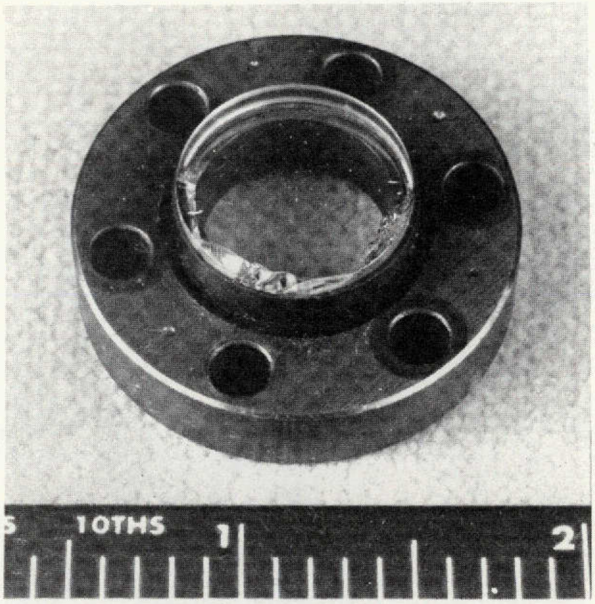

Figure 3A-5. Window Assembly Which Cracked during Mounting

In NASA Space Vehicle Design Criteria for "Buckling of Thin Walled Circular Cylinders" (Reference 3), the following equation is recommended for the buckling pressure, P, for a material with a Young's modulus E.

$$P = \frac{0.855}{\left(1 - \mu^2\right)^{0.75}} \frac{E\sqrt{\gamma}}{\left(\frac{r}{t}\right)^{2.5}\left(\frac{L}{r}\right)} \tag{1}$$

The value $\gamma = 1$ has been determined as the theoretical value; however, a correlation factor of $\sqrt{\gamma} = 0.75$ is recommended. For a Poisson's ratio of $\mu = 0.3$, the buckling pressure is then given by:

$$P = 0.69 \frac{E}{\left(\frac{r}{t}\right)^{2.5}\left(\frac{L}{r}\right)} \tag{2}$$

where r is the cylinder radius, L the length, and t the wall thickness.

According to this equation, the buckling pressure increases by a factor $n^{2.5}$ if the wall thickness-to-radius ratio increases by a factor n. Equation (1) has been compared with experiment, and design according to this equation can be considered to be conservative.

Figure 3A-6 shows the thermal conduction for a material with a conductivity of 2×10^{-3} W/m°K along a tube 25 mm long.

NASA has published a series of design criteria to be used as guides for design of space vehicles. One of these, Reference 4, entitled, "Buckling of Thin Walled Truncated Cones," gives as a lower bound for experimental data for unstiffened cones under hydrostatic loading:

$$P_{cr} = 0.75 \times 0.92E \left(\frac{\bar{\rho}}{L}\right)\left(\frac{t}{\bar{\rho}}\right)^{2.5} = 0.69E \left(\frac{\bar{\rho}}{L}\right)\left(\frac{t}{\bar{\rho}}\right)^{2.5} \tag{3}$$

where E is the Young's modulus of the cone material, L is the slant height, $\bar{\rho}$ is the average radius, and t is the shell thickness.

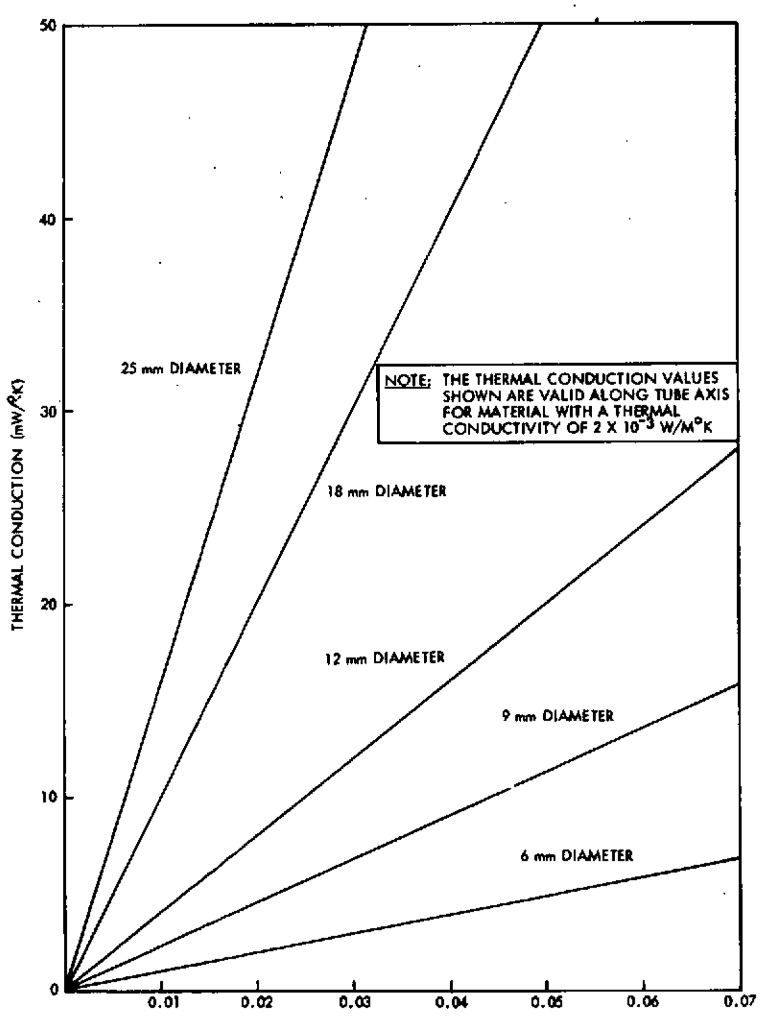

Figure 3A-6. Thermal Conduction of a Tube 25 mm Long

Reference 4 states that L, $\bar{\rho}$, and t define an "equivalent" cylinder which allows theoretical and experimental results for hydrostatic loading of cylinders to be applied to cones. This reference also cites Reference 5 as applying to ring-stiffened cones, but for lack of supporting data, does not recommend the use of the approximate buckling formulas presented there. These are given below

$$P_{cr} = 0.92E \left(\frac{\bar{\rho}}{L}\right)\left(\frac{t}{\bar{\rho}}\right)^{2.5} g(\psi) \left[(1+\eta_2)^{0.75} - \left(\frac{\bar{\rho}}{L}\right)\left(\frac{t}{\bar{\rho}}\right)^{0.5} \eta_2\right] \quad (4)$$

$$\eta_2 = \frac{12(1-\nu^2)\, I_{22}}{a_o t^3} + 12\left(\frac{\bar{Z}_2}{t}\right)^2$$

for closely spaced ring stiffeners. I_{22} is the moment of inertia of the stiffener cross-section, a_o is the stiffener spacing, ν is Poisson's ratio, and Z_2 is the offset of the centroid of the stiffened section from that of the skin.

It is apparent that the first of these equations contains the critical pressure for an unstiffened cone and a correction factor for stiffening. It is assumed that the same 0.75 factor which provides a lower bound to the data scatter band applies also. The stiffening correction factor F, then, is

$$F = g(\psi) \left[(1+\eta_2)^{0.75} - \left(\frac{\bar{\rho}}{L}\right)\left(\frac{t}{\bar{\rho}}\right)^{0.5} \eta_2\right] \quad (5)$$

$$\eta_2 = \frac{12(1-\nu^2)\, I_{22}}{a_o t^3} + 12\left(\frac{\bar{Z}_2}{t}\right)^2 \quad (6)$$

where $g(\psi)$ is a correction factor for the taper of the cone, and η_2 is a device for "smearing" the stiffener section over the skin area.

In Reference 3, a companion criteria document to Reference 4, Equation (17) of this reference is identical to the one given above, for unstiffened cones [Equation (3)], in line with the concept of an "equivalent" cylinder. It is of interest, then, to consider an "equivalent" stiffened cylinder as indicative of a stiffened cone.

For this purpose the following equations apply.

$$P_{cr} = \frac{5.513}{L\bar{\rho}^{1.5}} \left[\frac{\bar{D}_y^{\,3}\left(\bar{E}_x\bar{E}_y - \bar{E}_{xy}^{\,2}\right)}{\bar{E}_y} \right]^{1/4} \tag{7}$$

$$\bar{E}_x = \frac{Et}{1 - \nu^2} \tag{8}$$

$$\bar{E}_y = \frac{Et}{1 - \nu^2} + \frac{EA_r}{a_o} \tag{9}$$

$$\bar{E}_{xy} = \frac{\nu Et}{1 - \nu^2} \tag{10}$$

$$\bar{D}_y = \frac{Et^3}{12(1 - \nu^2)} + \frac{EI_{22}}{a_o} \tag{11}$$

where A_r is the cross-sectional area of the ring.

There is also the possibility that if the shell is made stable enough the shell material may yield in compression. The equation for this is:

$$P_{cy} = F_{cy}\left(\frac{t}{\bar{\rho}}\right) \tag{12}$$

where F_{cy} is the modulus of rupture. For the specific design which is shown in Figure 3A-7, certain simplifying approximations must be made because both the shell thickness and the ring spacing vary. The wall thickness increases toward the large end of the cone so that the ratio of thickness to radius does not vary much. Using an average value of this ratio and material properties for Inconel from Reference 6, the buckling failure load for this structure considered as an unstiffened cone is calculated from Equation (3)

$$P_{cr} = 1.63 \times 10^7 \text{ N/m}^2 \tag{13}$$

The buckling failure load for the stiffened configuration, considered either as a cone or an equivalent cylinder, exceeds the Venus surface requirements. The compressive yield strength calculated from Equation (12) is

$$P_{cy} = 4.90 \times 10^7 \text{ N/m}^2 \tag{14}$$

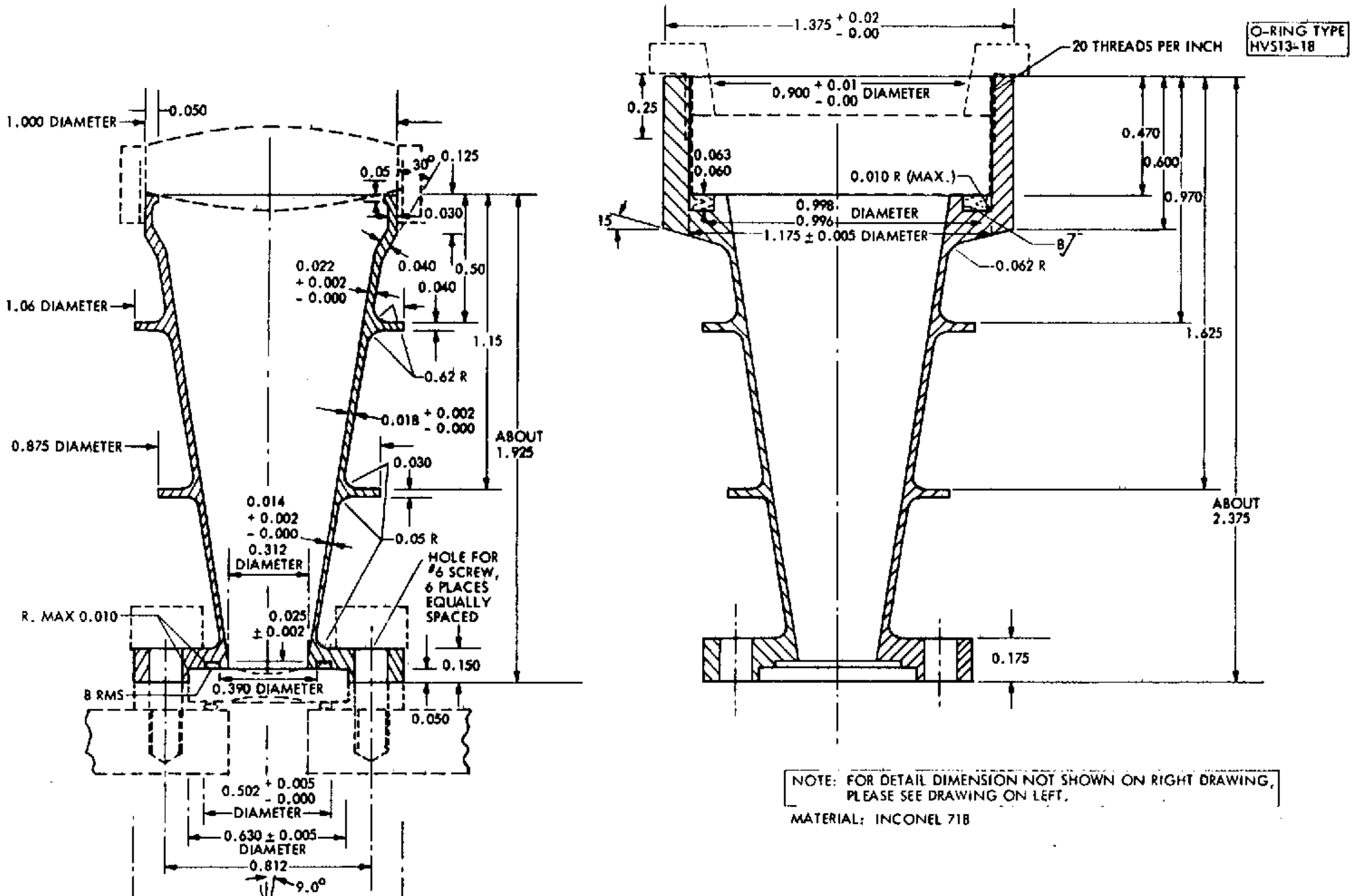

Figure 3A-7. Design of Conical Window Assembly

This figure is considerably in excess of the buckling failure figure. The design is reasonably adequate even when considered to be an unstiffened cone.

3. SEALS FOR VENUS PROBE WINDOWS

The major requirements for Venus probe window seals follow.

1) Minimum leakage is desired and worst-case leakage should not cause any significant increase in heat transfer, window contamination, or interference with the probe system.

2) The seal must be compatible with the probe environment including shock and vibration.

3) Low weight, low thermal conduction, and high reliability are other important characteristics.

The bonding of window material and support structure provides excellent seals and requires little additional weight. Sapphire can be wetted by glass, titanium, zirconium or molymanganese mistures. It can be matched to titanium, molybdenum, the high nickel-iron alloys such as Carpenter 49, copper-nickel alloys, Kovar, and the Corning glass 7520. Bonds can be made directly to Corning 7052. The seal shear strength of sapphire-Kovar joints is greater than 7×10^{7} N/m^{2} and leak rates are less than 10^{-15} standard cubic meters of helium per second. Braze systems for sapphire metal assemblies are available for service at 1873, 1673, 1273 and 1073^{o}K in vacuum, air, alkali metal environments and halogen atmospheres, respectively.

To achieve metal-to-window bonds, a close match between the coefficients of thermal expansion over the temperature range is important. Because this cannot be achieved completely, an elastic tubular support structure is needed. To avoid excessive stresses the tubing bonded to the window must be thin-walled. For sapphire windows with a diameter of 5 to 25 mm, a wall thickness of no more than 0.5 mm is generally used for Kovar or copper-nickel rings bonded to the sapphire window. Thin-wall 0.3 mm Kovar tubes are not strong enough to reliably withstand Venus surface conditions. Figure 3A-8 shows the design of a Kovar window assembly. At room temperature, it withstood more than 1.4×10^{7} N/m^{2} but collapsed at 9.3×10^{6} N/m^{2} and 725^{o}K (Figure 3A-9). Apparently the thin (0.38 mm) tube section next to the sapphire failed.

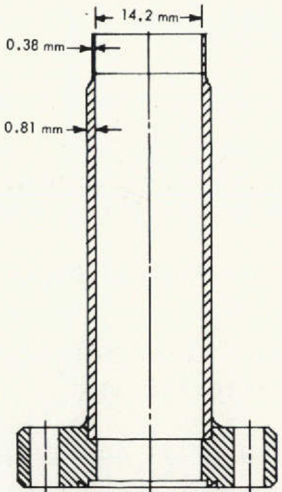

Figure 3A-8. Sapphire Window with Tubular Support Structure

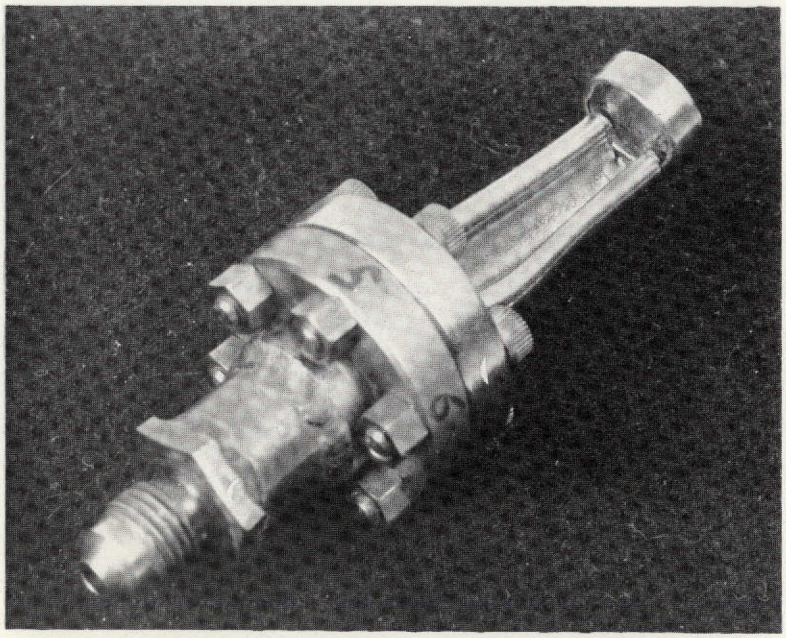

Figure 3A-9. Kovar Window Assembly after Collapse at 9.3×10^6 N/m^2 and 725°K

To achieve a nonmagnetic bond between sapphire and Inconel 718 the design shown in Figure 3A-10 was chosen after several less successful attempts. A ring of nonmagnetic copper-nickel alloy is copper brazed at about 1400°K to the metallized sapphire window. Then this unit is brazed at approximately 1100°K to the nickel-plated Inconel tube. It is desirable to braze at the end of the copper-nickel ring. The nickel plating is magnetic; however, more than 90 percent of the nickel film could be removed after brazing or the nickel plating could be replaced by nonmagnetic gold or silver plating. The basically nonmagnetic sapphire-to-Inconel 718 bond was successfully tested at 9.3×10^6 N/m^2 and 725°K. The leak rate was measured by sensing gas volume that leaked into a water container. No gas bubbles caused by leakage could be observed.

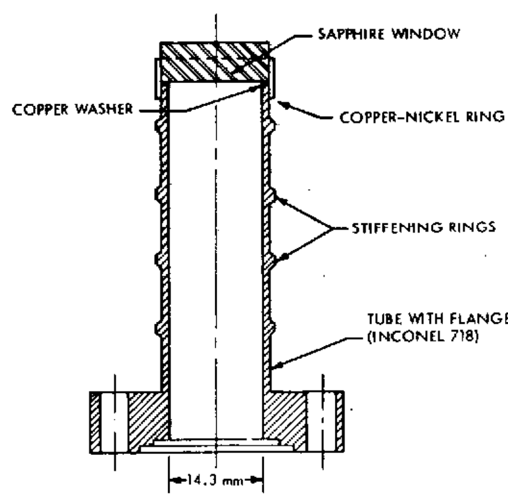

Figure 3A-10. Design of Inconel Window Assembly

Figure 3A-11 shows major tradeoff considerations between various types of high pressure seals. Weight is a critical parameter for space probe design and the weight penalty required for the seal is directly related to the force required to achieve a seal. Gaskets manufactured of asbestos with silicone or Viton, function over the temperature range of interest but have a large surface area and require high compressive forces. Metallic gaskets are frequently used in vacuum systems up to 720°K. The compressive forces are on the order of several hundred thousand Newtons per circular meter. When we heated such seals we experienced loose bolts after cooling. Apparently the material of the copper gasket flowed at the high temperatures (670°K).

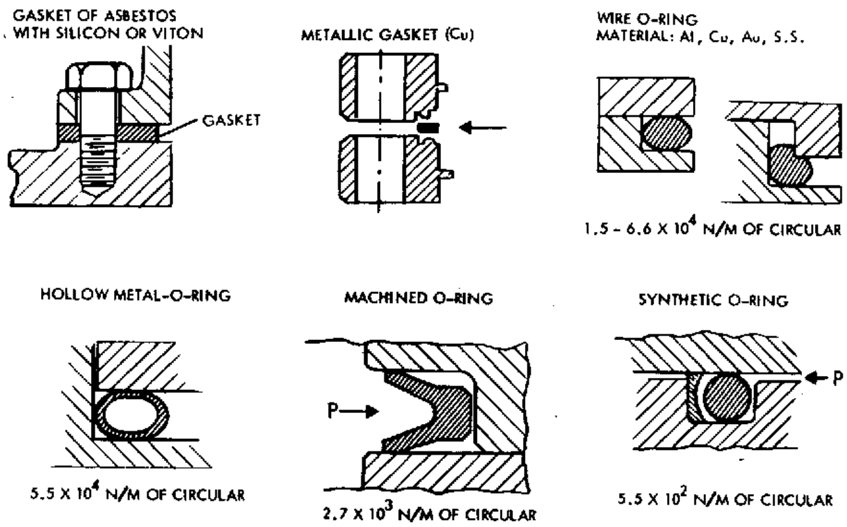

Figure 3A-11. Major Tradeoffs Between Seal Designs

Hollow metal O-rings were frequently used in our designs. Typical dimensions are 1.5 mm outside diameter of a silver-plated Inconel tube with a 0.25 mm thick wall. The ends of the tube are welded together to form a ring. At 1.4×10^7 N/m^2 and 700^{o}K typical CO_2 leakage was on the order of 10^{-5} std m^3 per hour. Teflon coating is also available for these O-rings.

Machined O-rings cost several times as much as hollow metal O-rings but we achieved good sealing at simulated Venus surface conditions. At 2.8×10^7 N/m^2 and 313^{o}K, or at 9.3×10^6 N/m^2 and 725^{o}K, we could not observe gas bubbles $(\leq 10^{-7}$ m$^3)$ escaping during a period of 300 seconds. Figure 3A-12 shows a test setup of four window assemblies in a Venus simulation chamber.

Preliminary tests with machined and hollow metal O-rings indicate that when vacuum is on the "high pressure side" and 1 atmosphere pressure is on the low pressure side (space flight condition), machined O-rings leak considerable more than hollow metal O-rings.

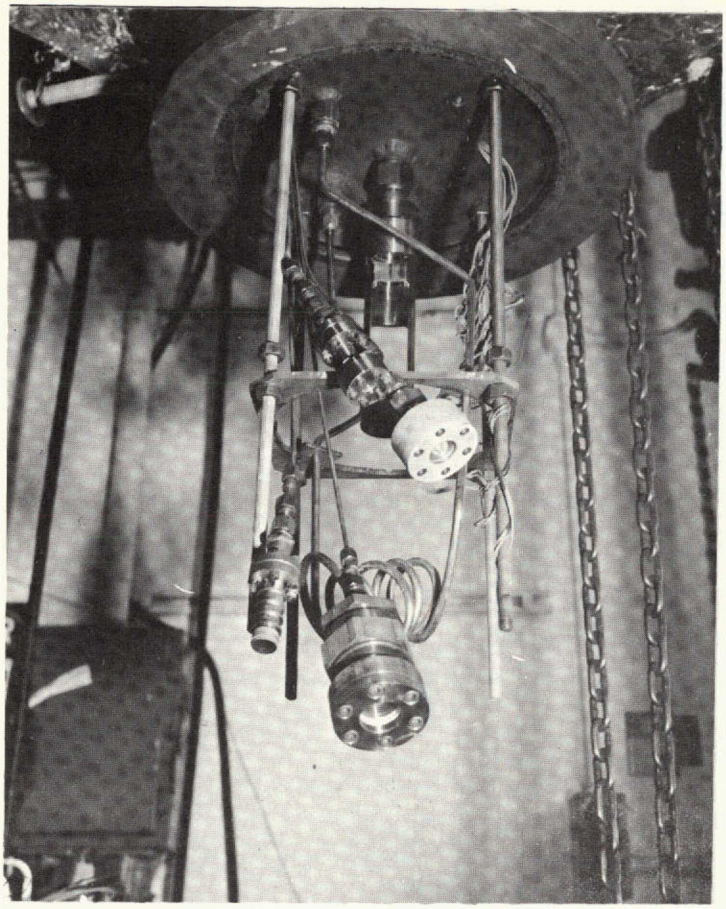

Figure 3A-12. Test Setup for Leak Test of Window Assemblies

4. WINDOW ASSEMBLY DESIGN

The tradeoff considerations in regard to structural materials, structural designs, and window sealing were combined to achieve optimum designs of window assemblies for Venus probes and/or Venus atmosphere simulation chambers. Figure 3A-13 shows the design of a sapphire window for our test chamber. The window aperture is 25 mm in diameter and all structural parts are made of stainless steel. The hollow O-rings provide not only a seal but also an elastic window support to avoid stress concentrations. The O-ring compression is limited by the window thickness and the additional gap allowed for the O-rings. The six bolts should be torqued systematically to 25, 50, and 100 percent of maximum torque. Figure 3A-14 is a photograph of this window assembly which was successfully tested at 9.3×10^6 N/m^2 and 725^oK.

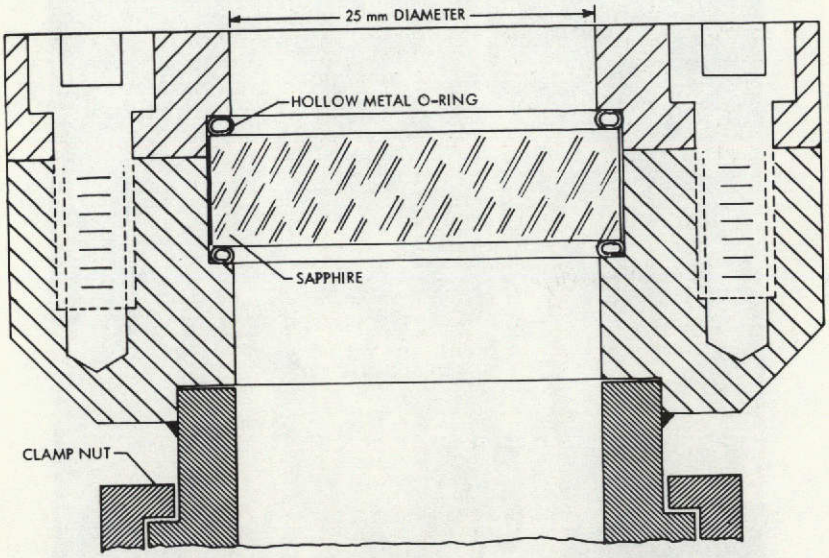

Figure 3A-13. Sapphire Window Design for Venus Atmosphere Simulation Chamber


Figure 3A-14. Completed Sapphire Window Tested Successfully at 9.3×10^6 N/m^2 and 725°K

During this test the design shown in Figure 3A-15 was also tested. The clamp ring and probe shell parts are made of aluminum. The six bolts are screwed into a helicoil. Because the sapphire window assembly is a double cylinder with diameters of 20 and 25 mm, a flat top surface is achieved. This would be an advantage if window wiping is desired.

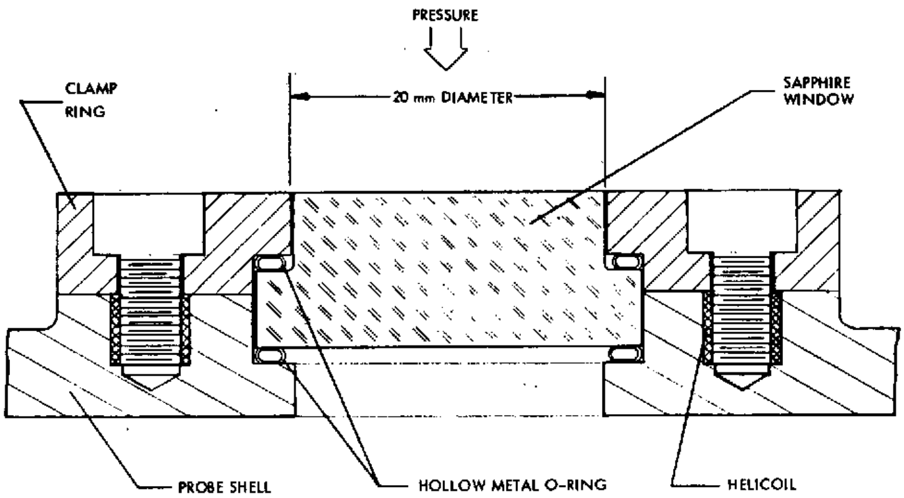

Figure 3A-15. High Pressure Window with Flat Top Surface

A major part of this effort was to design, build, and test sapphire and IRTRAN 2 windows that would be suitable for Venus probes and are nonmagnetic. Because it is doubtful that an IRTRAN-2-to-metal bond can be achieved, two approaches with clamped double windows were designed, manufactured, and tested. The first design is shown in Figure 3A-16. For the second design (Figure 3A-17), the weight of the assembly was considerably reduced; with the two windows it weighs approximately 0.1 kg. The machined O-rings sealed very well. At 725°K and 9.3×10^6 N/m^2, no CO_2 bubbles leaked through the IRTRAN 2 seal. For a sapphire window the heater was attached to a groove around the sapphire window port. In this manner, good thermal coupling between heater and window is achieved while the contacts to the window assembly structure have small Inconel cross sections and cause little heat conduction.

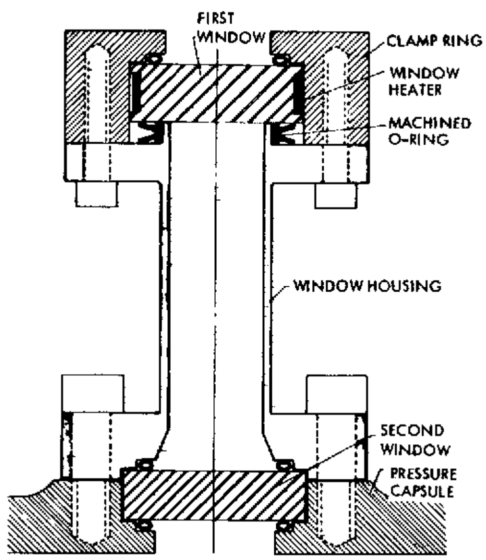

Figure 3A-16. Tubular Window Assembly with Two Clamping Seals

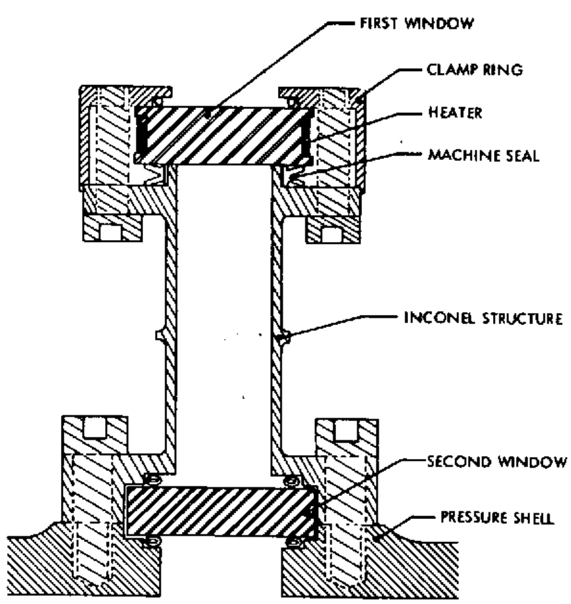

Figure 3A-17. Lightweight Design of Tubular Window with Compression Seals

In spite of the efforts to achieve a lightweight clamped window, this type of seal is heavier than a window assembly with metal-to-sapphire bonding. The bonded seal allows minimum weight for sapphire window and window seal. The design of the window bonded assembly is shown in Figure 3A-18. Two window assemblies of this type were manufactured. They have wall thicknesses of 0.76 and 0.62 mm which corresponds to calculated collapse pressures of 3.72×10^8 and 2.34×10^8 N/m^2. The window assembly shown in Figure 3A-19 was successfully tested at 725°K and 9.3×10^6 N/m^2. No leakage could be measured. Initial difficulties in achieving bonded seals between sapphire and Inconel 718 were overcome. This bonding technique will also be very useful for the conical window design described below.

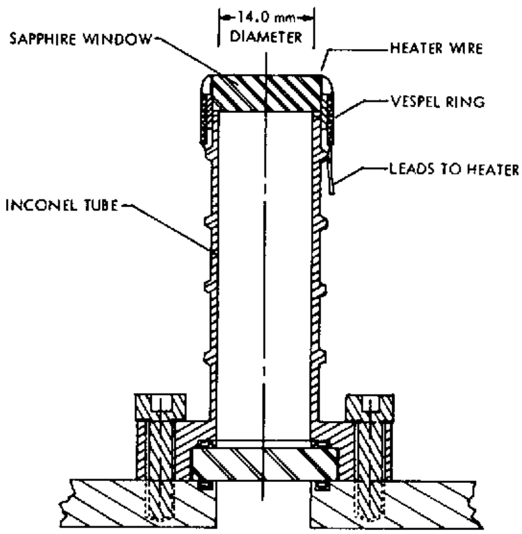

Figure 3A-18. Tubular Window Assembly with Heater and Sapphire Window Bonded to Metal

Cylindrical shells under hydrostatic outside pressure can often be designed with stiffening rings to be stress limited. The tube cross section and thermal conduction along the tube increase proportionally to the tube diameter. For buckling-limited cylinders, the wall thickness increases by approximately a factor $n^{0.6}$ when the diameter increases by a factor n. Therefore, the thermal conduction increases approximately proportional to $n^{1.6}$. For windows requiring a narrow field of view, a reduction of thermal conductance can be obtained without loss of aperture by constructing the window with a conical configuration having a smaller average diameter than that of its cylindrical equivalent.

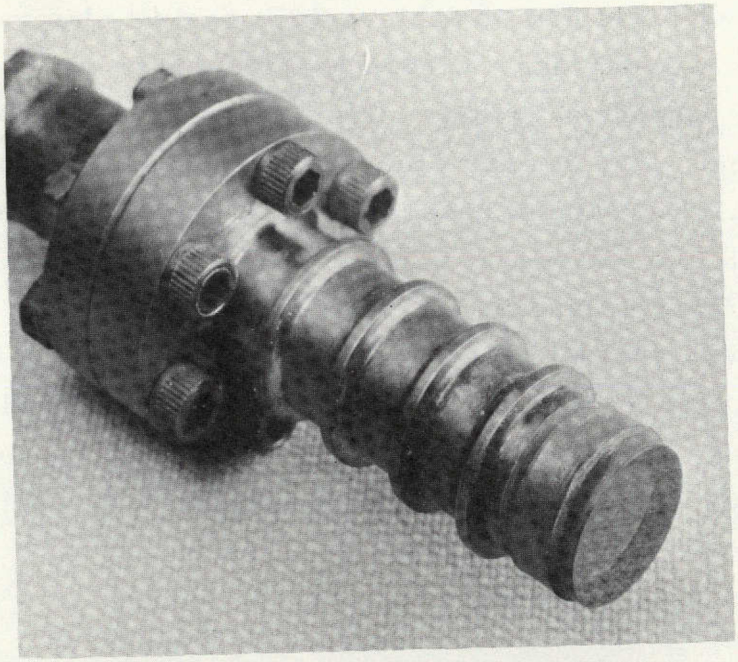

Figure 3A-19. Sapphire Window Bonded to Tubular Inconel Structure

Figure 3A-20 shows a cross section of a conical window; manufacturing details are shown in Figure 3A-7.

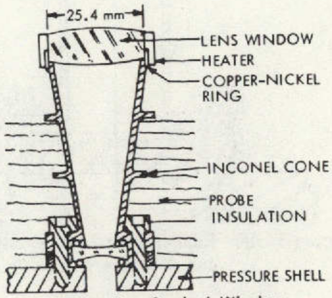

Figure 3A-20. Conical Window Mounted to Probe Wall

Only one stiffening ring is within the thermal insulation because it shortens the effective tube length in regard to thermal impedence. Six bolts are used to seal and mount the window assembly to the probe wall. The 25 mm diameter lens or window is bonded to the Inconel 718 cone in a manner similar to that used for the tubular window shown in Figure 3A-18.

Figures 3A-21 and 3A-22 are conceptual designs of viewing ports with 2π steradian field of view and diffusers for one version of a solar radiometer. These viewing ports are mounted near an antenna and therefore should not be conductive. Light from the diffuser is conducted by light pipes to the light detector. Internal reflection along the walls of the light pipe is caused either by a glass coating with a lower index of refraction or gas surrounding the light pipe. The light pipe is separated to achieve a high thermal impedence without losing a considerable portion of the light signal. Contact areas with the light pipe are kept to a minimum to avoid significant losses of light signal.

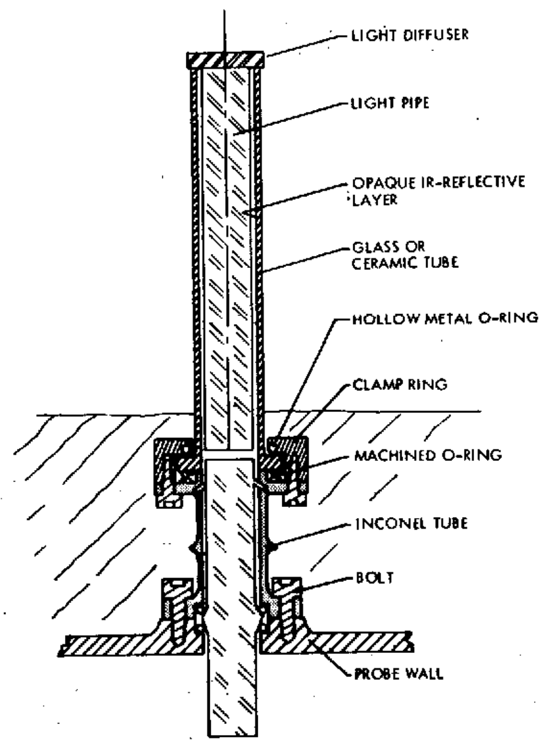

Figure 3A-21. Extended Viewing Port with Diffuser and Uncoated Light Pipe

Figure 3A-23 shows concepts for two window-wiping devices. One concept uses an electrical motor that is heat-sinked to the cool pressure shell. The mechanical motion is coupled to the wiper by means of a rod.

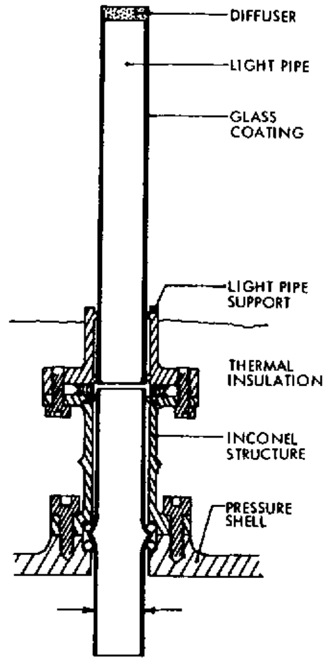

Figure 3A-22. Extended Viewing Port with Diffuser and Coated Light Pipe

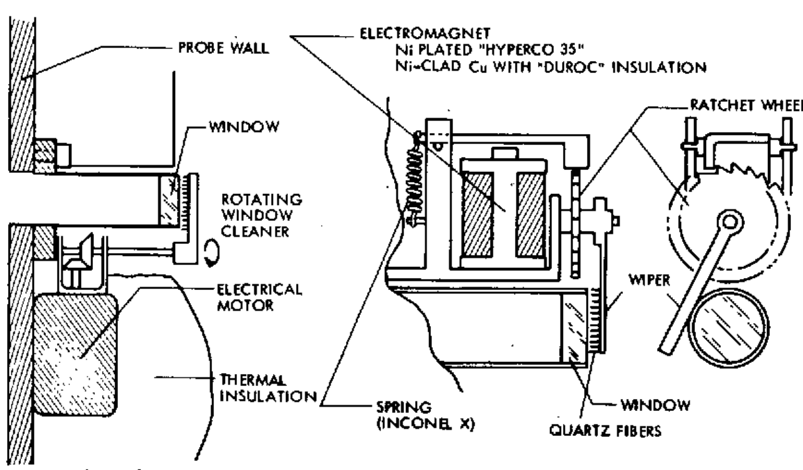

Figure 3A-23. Two Concepts for Electromechanical Drives to Wipe Windows

A simple ratchet drive similar to those used in stepping switches is less efficient than a motor, but the development cost is lower than that of a high temperature motor. A low-weight drive appears to be more important than efficiency for this purpose. We have demonstrated the feasibility of an electromagnet and a ratchet wheel drive at $750^{\circ}K$. Figure 3A-24 shows a concept to simulate cloud effects and particles to evaluate the performance of window assemblies. A postulated cloud compound is heated above ambient to generate vapor of the cloud compound. A cooler window assembly is positioned near this vapor. The effect of the vapor on the window transmittance is measured by means of a light source and a heated chamber window. The test setup allows injection of solid articles.

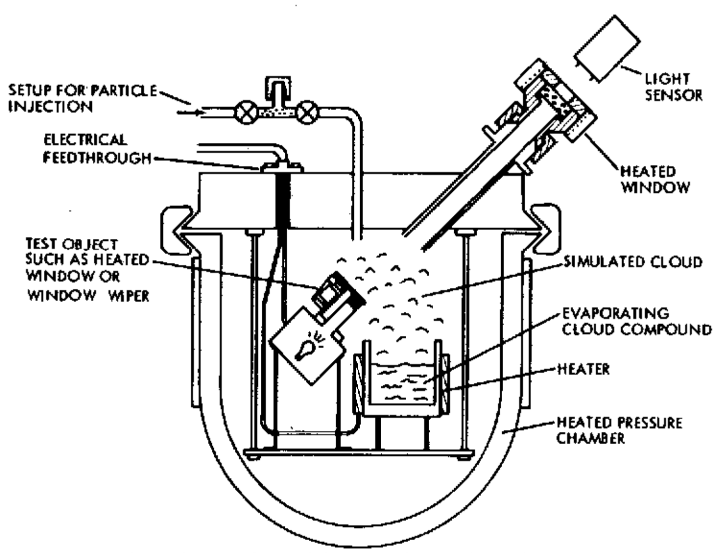

Figure 3A-24. Concept for Cloud/Dust Simulation and Evaluation of Window Assembly

5. REFERENCES

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2. "Kodak IRTRAN," Kodak Publication U-72, Eastman Kodak Company, Rochester, New York (1971).

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APPENDIX 3B

ACCOMMODATING THE MAGNETOMETER

APPENDIX 3B

ACCOMMODATING THE MAGNETOMETER

1. REMANENT MAGNETIC FIELD MODELING

A magnetic model of the Pioneer Venus small probe was generated. A computer program used the model to calculate expected remanent fields at the magnetometer sensor location shown in Figure 3B-1. The modeling was based on the following assumptions:

1) Field values of individual component parts were based on test data from the Pioneer Jupiter program for the demagnetized condition, and therefore the values calculated presume a magnetic control program of the same scope as used for Pioneer 10.

2) A dipole approximation (this may be optimistic) was used to calculate the change in field with distance from the geometrical center of the assembly to the geometrical center of the magnetometer sensor.

3) One half of the IC flat packs in an assembly were aligned parallel to the Y axis of the probe and one half were aligned parallel to the Z axis of the probe. Lead length for the IC's was set at a maximum of 0.33 cm.

4) In cases where no test data were available for a specific part, a field value was established based on a comparison with a similar device for which field data were available.

5) The computer program transforms the vector fields of the subsystem assemblies into probe coordinates and performs a vector summation at the location of the magnetometer sensor.

The resultant estimated remanent field at the location of the geometrical center of the sensor (as shown in Figure 3B-1) is 1114 nT. The sensor is assumed to be a single-axis fluxgate with axis oriented at 57 degrees from the small probe symmetry axis. (Equal field components along the X, Y, and Z axes would produce equal contributions in the sensor at this orientation). The component of the calculated remanent field parallel to the sensor axis would be 775 nT. Two alternative sensor locations resulted in total remanent fields of 1126 nT and 1670 nT.

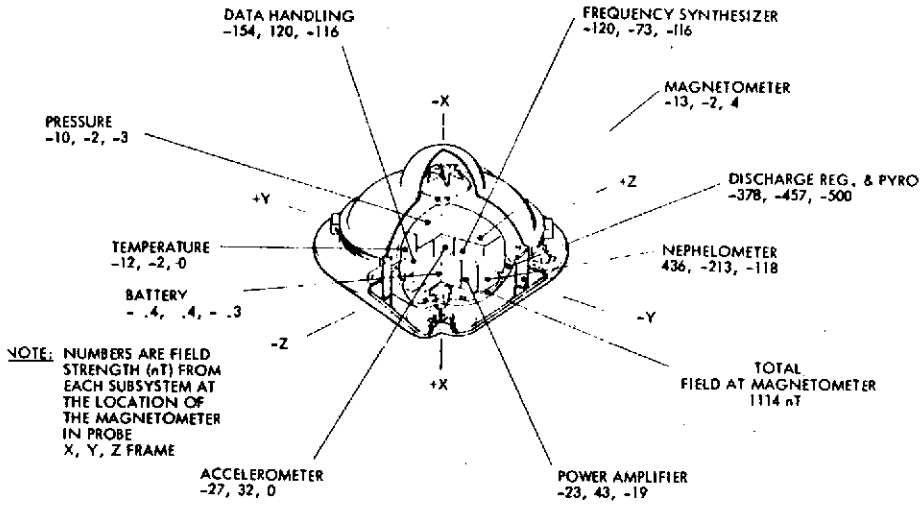

Figure 3B-1. Small Probe Magnetic Field Determination

To estimate the total field including stray fields, we have added a figure of 20 percent for stray fields. This increases the calculated field to approximately 1400 nT. This value for the stray field derives from analogy to Pioneer 10. On that spacecraft, the stray fields at the assembly level ran approximately 100 percent of the remanent field but at the spacecraft level, the stray field was virtually undetectable because of cancellations among the assemblies. The Pioneer Venus small probe will be packaged more densely, and will use lower powered currents so that a value of 20 percent appears reasonably conservative.

2. MAGNETIC CONTROL

The value of 1400 nT will vary depending on the relative positioning and orientation of the sensor and the assemblies that contribute the major share of the field. This variation is estimated to be 1400 nT $\pm$ 50 percent based upon a number of computer calculations and examination of those assemblies contributing the large portion of the field. Three ways by which the field at the sensor may be reduced are as follows:

1) Increase the separation distance between the sensor and the local field

2) Repackage the assemblies to stack together, obtaining a more efficient use of the separation effect and essentially eliminate stray fields by eliminating interconnecting cabling.

3) Eliminate a significant portion of the remanent field using hybrid electronics and DIPS.

All locations greater than 13 cm from the probe center are outside the pressure vessel in the Thor/Delta probe and expose the sensor to the ambient temperature. For the Atlas/Centaur the maximum distance is 19 cm. One type of flux gate has been shown to perform with no temperature drifts up to 338°K. A manufacturer of another type of flux gate sensor has indicated that sensors have been built to operate satisfactorily to 473°K and that it would probably not be too difficult to extend this range to 573°K with proper wire insulation and structural thermal compensation. However, the loss of permeability of suitable core materials becomes a serious problem in the 673°K range and would require a large metallurgical research program to produce a suitable alloy.

Without thermal protection, operation to 338°K would limit the measurements to altitudes above 51.5 km, while operation to 673°K would allow measurements to be taken down to approximately 12 km of the surface.

It appears that within the present size of the small probe, it is not possible to keep the field below 100 nT by magnetic cleanliness design and control. The lowest value achievable appears to be approximately 630 nT, and to achieve this requires a magnetic control program such as Pioneer 10 at a cost of $500 000 and a new integrated packaging concept as shown on Figures 3B-2 and 3B-3. In keeping with the cost saving philosophy of this program, it may be possible to show some significant savings in the cost of magnetic control and keep the program at Pioneer 10 cleanliness.

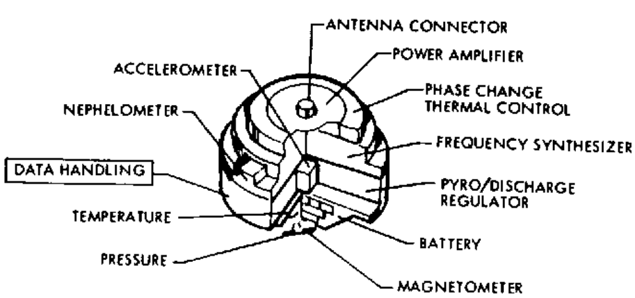

Figure 3B-2. Integrated Electronic Package Concept for Thor/Delta

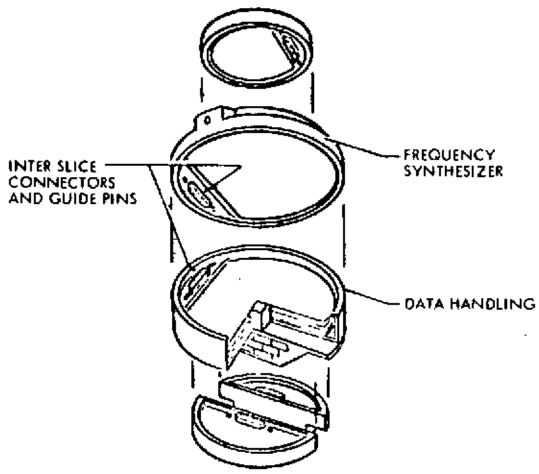

Figure 3B-3. Thor/Delta Small Probe Electronics Interconnection

A 19-cm separation distance can be obtained inside the Atlas/Centaur probe. At this distance the field could be limited to 295 nT using Pioneer 10 magnetic cleanliness technology.

To approach a value of 100 nT at the sensor, we must use a magnetometer developed for high temperature operation. Locating the sensor within the aeroshell (as in Figure 3B-4) at a maximum separation distance from the Thor/Delta launched probe results in a field level of 46 nT. This approach involves the same magnetic control cost as discussed above, plus the use of hybrid electronics and dual in-line packages (DIP's) for some of the assemblies at a cost of $250 000, plus the cost of developing a high temperature sensor, plus the cost of developing thermal insulation that might be used to further decrease the altitude to which the sensor will operate. Obviously these costs will depend upon the altitude to which operation is required. Furthermore, both the sensor and the insulation will add weight (~0.4 kg) to limit sensor temperature to 583°K at the surface (see Section 3. below) over that for a sensor accommodated inside the pressure vessel. Part of the development costs and weight will be associated with the wires and connectors between the remotely located sensor and its electronics located within the pressure vessel. In addition, the temperature of the sensor must be accurately monitored by a thermistor or thermocouple (additional cost and weight) and these measurements included in the data stream.

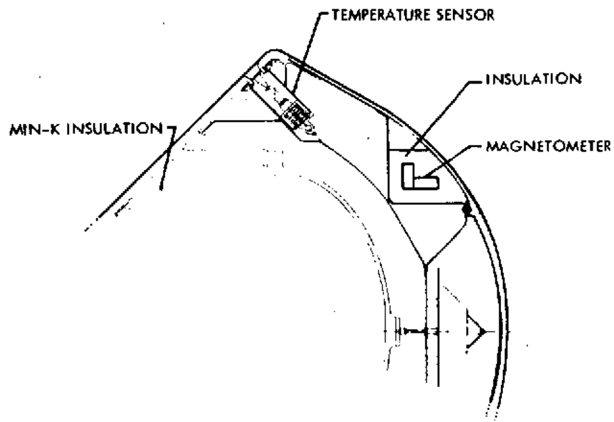

Figure 3B-4. Triaxial Fluxgate Sensor Located Within Aeroshell on Thor/Delta Small Probe

A fixed sensor located outside the aeroshell was initially considered but dropped because of the perturbations it could induce during probe entry into the atmosphere. The next category considered was a sensor stowed within the aeroshell during entry and then deployed 7 to 10 cm beyond the base cover to a distance of 30.5 cm from the probe center. Two mechanisms were envisioned depending upon the dimensions of the magnetometer in its thermal protection envelope. If the dimensions are no greater than 4 to 5 cm in diameter, then a mechanism analogous to the PAET temperature probe deployment could be used. If the dimensions exceeded 5 cm diameter, then a mechanism of the type shown in Figure 3B-5 was considered. In either case, two and preferably three such devices were necessary to balance the aerodynamic effects of the projecting mass and surface. Very preliminary estimates of the program cost implications of this deployment are $300 000.

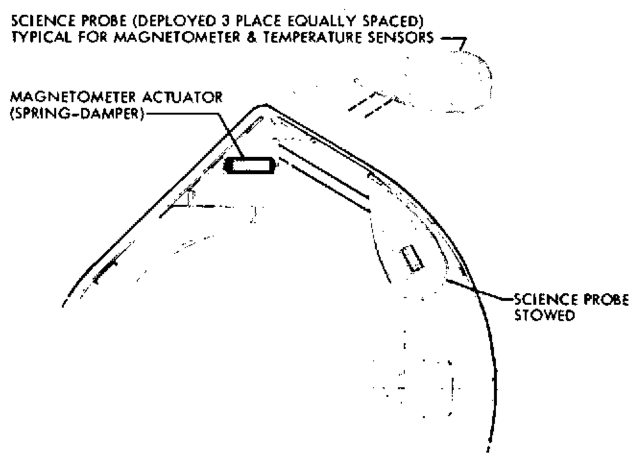

Figure 3B-5. Deployable Biaxial Fluxgate Sensor on Thor/Delta Probe

To summarize the above considerations, four locations for the magnetometer sensor are illustrated in Figure 3B-6. These locations, designed to reduce background magnetic field seen by the sensors are:

1) Sensor located within the pressure vessel at maximum separation from subsystems with high remanent field.

2) Temperature-protected sensor located within the aeroshell at a maximum distance from the pressure vessel.

3) Temperature-protected sensor located on short boom deployed through base cover opening (30 cm from probe center) after entry with two dummy sensors to balance configuration.

4) Temperature-protected sensor located on tripod (self-erecting after entry) with sensor along roll axis (63 cm from probe center).

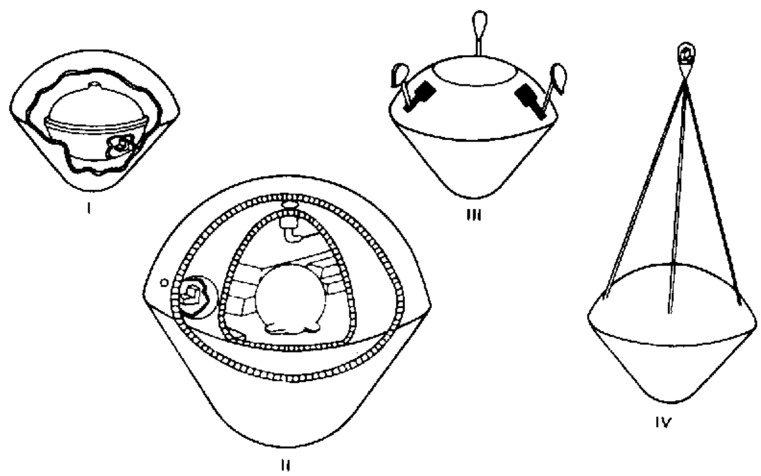

Figure 3B-6. Magnetometer Accommodation Concepts

The magnetic cleanliness impact of these configurations are itemized in Table 3B-1. The table shows the estimates of the magnetic field background at each magnetometer sensor location for four levels of magnetic cleanliness.

Table 3B-1. Impact of Magnetic Control on Thor/Delta Launch Design

LOCATION	MAGNETIC CONTROL LEVEL 1		LEVEL 2		LEVEL 3		LEVEL 4	
I	630 nT	$ 500 000 0 KG	405 nT	$ 750 000 0 KG	7 000 nT	$100 000 0 KG	14 000 nT	$0 0 KG
II	295 nT	$ 700 000 0.4 KG	46 nT	$ 950 000 0.4 KG	1 400 nT	$300 000 0.4 KG	2 800 nT	$200 000 0.4 KG
III	50 nT	$1 000 000 1.5 KG	10 nT	$1 250 000 1.5 KG	500 nT	$600 000 1.5 KG		
IV					55 nT	$600 000 1 KG	110 nT	$500 000 1 KG

LOCATIONS ARE IDENTIFIED IN FIGURE 3A-6. THE MAGNETIC CONTROL LEVELS ARE:

LEVEL 1 - PIONEER 10 MAGNETIC CLEANLINESS

LEVEL 2 - PIONEER 10 MAGNETIC CLEANLINESS PLUS THE USE OF HYBRIDS AND DIP TO REDUCE REMANENT FIELD FROM LEADS AND CASES

LEVEL 3 - REDUCED LEVEL OF MAGNETIC CLEANLINESS SUCH AS EMPLOYED ON PARTICLES AND FIELDS SUBSATELLITE

LEVEL 4 - NO FORMAL MAGNETIC CLEANLINESS BUT GOOD DESIGN AND PROCUREMENT PRACTICES

The total program cost elements for design, development and fabrication are as follows:

Magnetic control
- Level 3 $100 000
- Level 1 $500 000
- Level 2
  - $500 000 magnetic cleanliness
  - $250 000 hybrids and DIP

High temperature sensor (580^{o}K) - $100 000

Temperature protection to hold sensor to 580^{o}K - $100 000

Deployable boom - $300 000

Deployable aerofin - $300 000

Planet reference receiver and antenna - $250 000

The weight elements are as follows:

Temperature protected sensor housing - 0.4 kg (per probe)

Three stub booms deployed through base cover - 1.1 kg (per probe)

One self-erecting boom on outside of base cover - 0.6 kg (per probe)

3. THERMAL ANALYSIS OF EXTERNALLY MOUNTED VENUS PROBE MAGNETOMETER

Four configurations for thermal protection of an externally mounted small probe magnetometer have been analyzed. The first configuration shown in Figure 3B-7 consisted of a two-element magnetometer imbedded in a 4.5-cm-diameter sphere of MIN-K insulation. The second configuration consisted of a 6.4-mm-thick spherical shell of water having an internal diameter of 4.5-cm surrounded by a 1-cm-thick spherical shell of MIN-K with an internal diameter of 5.7 cm. The third configuration consisted of a 1.6-cm-thick spherical shell of water having a 4.5 cm inside diameter surrounded by a 1.3-cm-thick spherical shell of MIN-K with an internal diameter of 7.7 cm. The fourth configuration consisted of the 4.5-cm-diameter sphere of configuration 1 inserted in the 4.5-cm-diameter internal cavity of configuration 2.

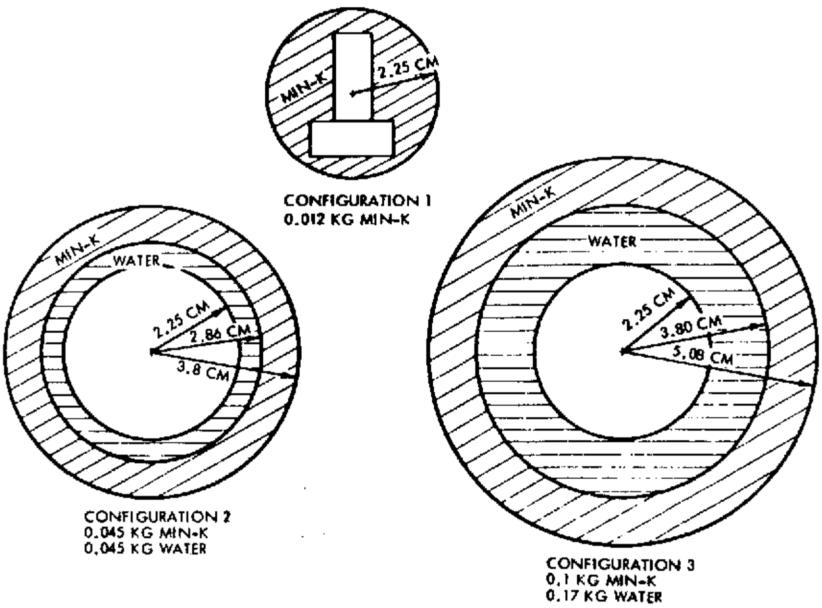

Figure 3B-7. **Magnetometer Sensor Thermal Protection Configurations**

Thermal analyses were performed using the temperature, pressure, composition, density, and viscosity corresponding to the SP-8011 Venus atmospheric model. The atmospheric properties were related to the time of descent via the small probe ballistic coefficient of 198 kg/m^2.

Figure 3B-8 shows the atmospheric temperature as a function of descent time. A computer program incorporating all of the atmospheric properties and a thermal network of the magnetometer shown in configuration 1 of Figure 3B-7 was used to calculate the instrument temperature as a function of time. This temperature response, shown in Figure 3B-8, makes it evident that no reasonable amount of passive insulation could maintain the temperature of an externally mounted magnetometer within the allowable limits.

Some attention was given to possible heat sink materials. Configurations 2 and 3 of Figure 3B-7 were analyzed using water as a heat sink material. The temperature response for each of these configurations was calculated manually under the assumption that the external insulation temperature was equal to the local atmospheric temperature. Thermal capacitances of both the MIN-K insulation and the liquid water phase were included; the internal 4.5-cm-diameter cavity was considered void.

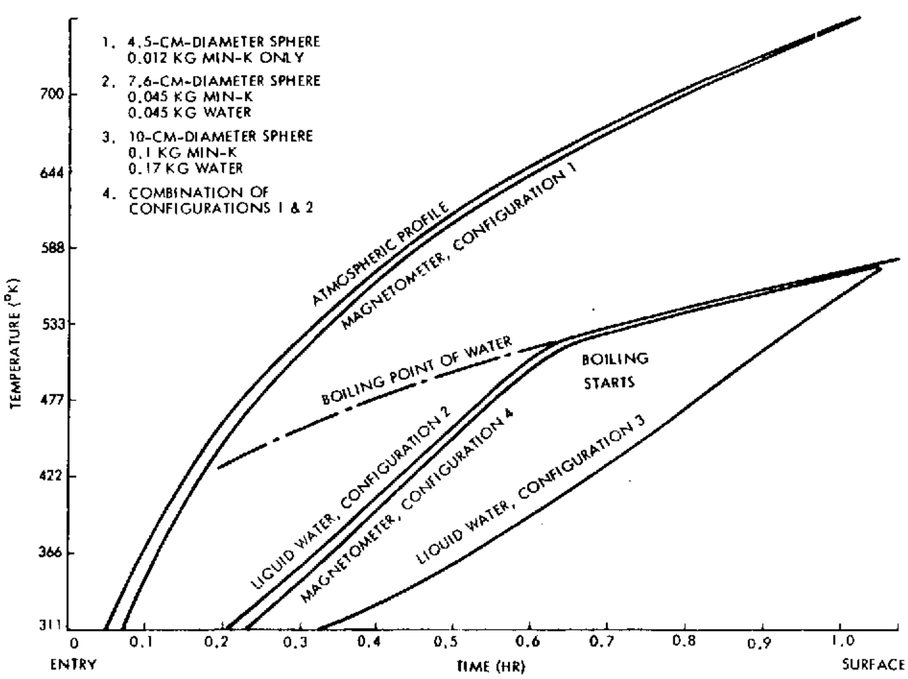

Figure 3B-8. Magnetometer Temperatures

For configuration 2, the water temperature reached the local boiling point at about 0.65 hours into the descent and essentially all of the 0.020 kg (0.045 pounds) of water had evaporated upon arrival at the planet surface.

For configuration 3, the liquid water temperature did not reach the local boiling point until the probe arrived at the planet surface. The thermal capacitance of the 0.075 kg (0.165 pounds) of liquid water was sufficient to absorb the energy transferred from the atmosphere without evaporation.

In configuration 4, the computer program for configuration 1 was used with the water temperature profile for configuration 2 substituted for the atmospheric temperature profile. The temperature response for the combined configuration shown as Configuration 4 in Figure 3B-8 indicates that the thermal protection offered by the 4.5-cm sphere of MIN-K around the magnetometer is insignificant relative to the water jacket.

Lower boiling temperatures could be achieved by substituting ammonia for water as the boiling heat sink material. The ammonia critical temperature is $405^{\circ}K$ at a critical pressure of 1.13×10^{7} N/m^{2}. The lower boiling point of ammonia relative to water is partially compromised by a lower latent heat $(1.37 \times 10^{6}$ J/kg$)$, a lower density $(600$ kg$/m^{3}$ at $297^{\circ}K)$ and a higher vapor pressure $(10^{6}$ N/m^{2} at $298^{\circ}K)$. In addition, the latent heat would decrease drastically as the critical pressure is approached near the planet surface.

Another possible heat sink candidate is ammonium carbonate, which has a pressure independent endothermic heat of decomposition of about 1.79×10^{6} J/kg at $333^{\circ}K$. The products of decomposition are all gases (ammonia, carbon dioxide, water vapor) so that no solid residue remains. Because the reaction is irreversible, premature decomposition must be prevented.

4. PLANET REFERENCE

Discussions with one of the magnetometer experimenters have indicated that the experiment would still be useful without a planet reference. His order of desirability for this information is:

1) An on-board sun sensor that pulses a biaxial sensor to take data at known solar azimuth angles

2) A reference received at earth, which can be used to deduce the magnetometer sensor axes positions at the time of each measurement

3) No basis for obtaining a reference of the planet direction when the magnetometer measurements were made.

Techniques to satisfy items 1 and 2 that have been considered to varying degrees are:

1) Antenna notch or spike (2)

2) Polarization pattern (2)

3) Modify nephelometer to detect sun (1, 2)

4) Add sun sensor at visible wavelengths (1, 2)

5) Add sun sensor at radio wavelengths (1, 2)

6) Detect uplink with directional antenna-receiver (1, 2).

Techniques 1) and 2) do not appear practical because there is an unacceptable communication penalty to provide a recognizable notch, spike, or polarization pattern in the downlink. This penalty, estimated at 3 dB, would bring the signal down below the margin for adverse conditions in the link, thereby compromising the entire experiment complement of the small probe. Technique 3) would complicate only the nephelometer. The nephelometer light source will probably be highly collimated to reduce the falloff in sensitivity with distance. The detector, however, could have a larger fan-shaped field of view, which would be sure to include the sun. Chopping the light signal from the nephelometer source would provide a basis for separating it from the dc solar signal. It is estimated that the modification to the nephelometer would increase its cost and weight by 10 and 20 percent, respectively.

Technique 4) would probably not be more expensive than modifying the nephelometer but could be two or three times heavier and would require another window. Both 3) and 4) would cease to be usable when the probe descended through a sufficient thickness of clouds to render the sun undistinguishable as a source. This could occur fairly high in the atmosphere so the utility of these techniques is significantly limited.

The possibility of using the sun as a directional radio source that could be viewed through the clouds and Venus atmosphere all the way to the surface was considered. The solar energy emitted during solar quiet periods at S-band is 1.9×10^{-20} W/cm^2 μm, and at X-band it is 2.9×10^{-18} W/cm^2 μm.

If we consider a superheterodyne receiver with a 5 kHz bandwidth, then the detectable energy fluxes would be 5.9×10^{-21} W/cm^2 at S-band and 6.4×10^{-29} W/cm^2 at X-band. To determine whether these fluxes are reasonable one must consider antenna size as it relates to detector noise levels for these fluxes. A further consideration must be given to antenna size in relation to the gain, and therefore directionality of the signal. A low-gain antenna does not have such directionality. Thus, for example, the antenna area required by the diffraction limit to obtain unity gain at S-band is 24.2 cm^2, whereas this area can provide a gain of 15 (11.8 dB) at X-band.

At this point we look at the S-band signal strength arriving at Venus from the 400 kW S-band uplink from the DSN transmitter. This signal is 2.16×10^{-16} W/cm^2, which is 37 000 times as intense as the solar S-band signal. It therefore appears that at S-band it is preferable to use the DSN as a directional source rather than the sun. However, we must now consider what size antenna is necessary to get a reasonably sharp signal pulse. Thus an antenna beam width of 28 x 27 degrees is obtained with antenna area of 34.5 x 45.6 cm = 1570 cm^2, which has a gain of 65 (18.1 dB). At S-band the antenna size and weight is limited by angular resolutions rather than by signal strength from the DSN. The above antenna, which is 23.9 cm deep and weighs 2.3 kg, is too large and heavy to be practical for the small probe. The alternative use of a unity gain antenna appears more reasonable in weight (0.11 kg), but would give a long pulse with a very poorly defined maximum. The receiver output would probably require processing to determine the angle at which the signal peaked. A small superhetrodyne receiver weighing 0.34 to 0.45 kg would be adequate. The total weight would then be 0.45 to 0.56 kg. The cost of this technique has been roughly estimated at $250 000 plus the cost of additional signal processing logic to be tied to the magnetometer if the output were used onboard to trigger the magnetometer (experimenter's preference 1). The receiver output could instead be sent back to earth and processed to yield planet reference data to determine on the ground where the sensor was pointing when the measurement was made (experimenter's preference 2). This would increase the data stream from the small probe, but there are probably no major cost differences using the technique in either mode. It may be very likely that ground processing of the signal would yield a more accurate knowledge of the sensor orientation than on-board processing.

The alternative of using the X-band solar signal has not been fully evaluated and might still be considered as a viable choice. The antenna size and weight required for 15 dB gain at X-band (sufficient to give good angular resolution and thereby satisfy experimenter's preference 1) is similar to the 0.11 kg, unity gain S-band antenna identified above. The viability will depend on whether a small X-band receiver can be designed for reasonable signal-to-noise ratio with an input signal of 0.8×10^{-18} watts (-151 dBm). This signal strength assumes a 24.2 cm^2 antenna (15 dB gain at X-band) and a 3 dB, attenuation in the Venus atmosphere.

5. PROBE SPIN

A spinning probe permits the experiment to be performed with a two-axis sensor. It, however, does not lower the data rate since the spinning axis must be read out with sufficient frequency to resolve the vector in that plane. If the probe spins it provides a basis for subtracting out the probe field component in the plane perpendicular to the spin from the planetary field. The probe field will be a DC signal (perhaps slowly varying in time) and the Venus field should be a modulation of this signal at the spin frequency. This technique of continuous calibration is useful only if a large number of measurements are taken over each spin and the azimuthal position of the probe is known in planet coordinates for each measurement. Alternatively, a measurement could be taken, say, once every spin but 60 degrees further in azimuth so that over five spins six measurements would be obtained over 360 degrees. To do this the planet reference signal triggering the measurement must be quite accurate and have more logic in it.

Spin and planet reference are therefore related in that there seems to be little value in having the probe spin unless a planet reference is also available. One of the experimenters has indicated that spin, like planet reference, is not a requirement of the experiment. If the probe were not spinning, a three-axis sensor would be used at a very slight weight penalty (~30 grams). Although the small probe will enter spinning, that spin will in all likelihood not be maintained during terminal descent. Some rate may be maintained and asymmetries in the surface of the ablator after entry may also produce some spin. Providing a specific spin rate would require deployment of fins after entry at a cost and weight penalty comparable to that for the magnetometer doors ($300 000), although there is some possibility that the spin surfaces could be combined with the doors.

6. CONCLUSIONS

- Design and control alone for magnetic cleanliness is not sufficient to keep the background field at the sensor down to 100 nT.

- For the small probe launched on Thor/Delta, achieving 100 nT requires development of a high-temperature sensor, use of hybrid electronics and DIP, and stringent magentic cleanliness at a total cost increment of close to $1 000 000.

- For the small probe configuration launched on Atlas/Centaur, it may be possible to save some of these costs by either eliminating magnetic cleanliness and using a high-temperature sensor on a boom or by mounting a low-temperature sensor inside the probe.

- Planet reference data for the magnetometer may be obtained to the surface by use of a separate radiometer detecting the DSN up-link at a cost of $250 000. A considerably cheaper technique would modify the small probe nephelometer to detect the sun. This would only be usable above the clouds.

- Providing positive spin control on the small probe would cost approximately $300 000. Part of this cost could be eliminated if the spin surfaces were made integral with the doors in the base cover through which the sensor was deployed.

SECTION 6 APPENDICES

APPENDIX 6A

COMMAND LIST
LARGE PROBE

APPENDIX 6A

COMMAND LIST

LARGE PROBE

	TO	FROM
SCIENCE		
Temperature Gauge Power ON	PCU	PCU/Probe Bus/EGSE
Pressure Gauge Power ON	PCU	PCU/Probe Bus/EGSE
Accelerometer Power ON	PCU	PCU/Probe Bus/EGSE
Neutral Mass Spectrometer Power ON	PCU	PCU/Probe Bus/EGSE
Cloud Particle Size Analyzer Power ON	PCU	PCU/Probe Bus/EGSE
Solar Radiometer Power ON	PCU	PCU/Probe Bus/EGSE
IR Flux Radiometer Power ON	PCU	PCU/Probe Bus/EGSE
Gas Chromatograph Power ON	PCU	PCU/Probe Bus/EGSE
Wind/Altitude Radar Power ON	PCU	PCU/Probe Bus/EGSE
Hygrometer Power ON	PCU	PCU/Probe Bus/EGSE
All Science Power ON/OFF	PCU	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 1	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 2	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 3	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 4	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 5	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 6	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 7	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 8	Mass Spectrometer	PCU/Probe Bus/EGSE

	TO	FROM
Mass Spectrometer Pyro Fire No. 9	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 10	Mass Spectrometer	PCU/Probe Bus/EGSE
Mass Spectrometer Pyro Fire No. 11	Mass Spectrometer	PCU/Probe Bus/EGSE
DHC		
Format A	DTU	PCU/Probe Bus/EGSE
Format B	DTU	PCU/Probe Bus/EGSE
Format D_1	DTU	PCU/Probe Bus/EGSE
Format D_2	DTU	PCU/Probe Bus/EGSE
DHC Subsystem Power ON/OFF	PCU	PCU/Probe Bus/EGSE
G-Switch Safe/Arm	PCU	PCU/Probe Bus/EGSE
Coast Timer Power ON/OFF	PCU	PCU/Probe Bus/EGSE
Coast Timer Reset	Coast Timer	PCU/Probe Bus/EGSE
Coast Timer Normal/ Accelerate	Coast Timer	Probe Bus/EGSE
DTU Bit Rate Normal/ Accelerate	Descent Timer	Probe Bus/EGSE
POWER		
Pyrotechnic Safe/Arm	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 1 - Mortar fire	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 2 - Mortar fire	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 3 - Aeroshell jettison	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 4 - Aeroshell jettison	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 5 - Afterbody parachute	Pyro Firing Circuits	PCU/Probe Bus/EGSE
Pyrotechnic No. 6 - Afterbody parachute	Pyro Firing Circuits	PCU/Probe Bus/EGSE

	TO	FROM
COMMUNICATIONS		
Communications Subsystem Power ON/OFF	PCU	PCU/Probe Bus/EGSE
S-Band Power Amplifier Power ON/OFF	PCU	PCU/Probe Bus/EGSE
THERMAL CONTROL	To	From
Window Heaters Power ON/OFF	PCU	PCU/Probe Bus/EGSE
Mass Spectrometer Heater Power ON-HIGH	PCU	PCU/Probe Bus/EGSE
Mass Spectrometer Heater Power ON-LOW	PCU	PCU/Probe Bus/EGSE

SMALL PROBE

	TO	FROM
SCIENCE		
Temperature Gauge Power ON	PCU	PCU/Probe Bus/EGSE
Pressure Gauge Power ON	PCU	PCU/Probe Bus/EGSE
Nephelometer Power ON	PCU	PCU/Probe Bus/EGSE
Accelerometer Power ON	PCU	PCU/Probe Bus/EGSE
IR Flux Detector Power ON	PCU	PCU/Probe Bus/EGSE
All Science ON	PCU	PCU/Probe Bus/EGSE
All Science OFF	PCU	PCU/Probe Bus/EGSE
DHC		
Format A	DTU	PCU/Probe Bus/EGSE
Format B	DTU	PCU/Probe Bus/EGSE
Format D	DTU	PCU/Probe Bus/EGSE
DHC Subsystem Power ON/OFF	PCU	PCU/Probe Bus/EGSE
G-Switch Safe/Arm	PCU	PCU/Probe Bus/EGSE
Coast Timer Power ON/OFF	Coast Timer	Probe Bus/EGSE
Coast Timer Reset	Coast Timer	Probe Bus/EGSE
Coast Timer Normal/ Accelerate	Coast Timer	Probe Bus/EGSE
DTU Bit Rate Normal/ Accelerate	Descent Timer	Probe Bus/EGSE
POWER		
Pin Puller Safe/Arm	PCU	PCU/Probe Bus/EGSE
Pin Puller No. 1 Initiate	PCU	PCU/Probe Bus/EGSE
Pin Puller No. 1 Backup Initiate	PCU	PCU/Probe Bus/EGSE
Pin Puller No. 2 Initiate	PCU	PCU/Probe Bus/EGSE
Pin Puller No. 2 Backup Initiate	PCU	PCU/Probe Bus/EGSE
Battery Heater ON/OFF	PCU	PCU/Probe Bus/EGSE

	TO	FROM
COMMUNICATIONS		
Communication Subsystem Power ON/OFF	PCU	PCU/Probe Bus/EGSE
S-Band Power Amplifier Power ON/OFF	PCU	PCU/Probe Bus/EGSE
THERMAL CONTROL		
Window Heaters Power ON/OFF	PCU	PCU/Probe Bus/EGSE

APPENDIX 6B

SCIENCE INSTRUMENT TELEMETRY SIGNAL CHARACTERISTICS

APPENDIX 6B

SCIENCE INSTRUMENT TELEMETRY SIGNAL CHARACTERISTICS

The science instrument telemetry signal characteristics are summarized in Tables 6B-1, 6B-2, 6B-3, and 6B-4.

Table 6B-1. Science Instrument Telemetry Signal Characteristics Large Probe

INSTRUMENT	CHARACTERISTICS	OPERATING ALTITUDE	BITS/ SAMPLE	MINIMUM DATA ACQUISITION RATES (SAMPLES/SECOND) CALI-BRATION	PRE-ENTRY	ENTRY	BLACK-OUT	POST-BLACK-OUT	TERMINAL DESCENT	POST-IMPACT	LOCATION SENSOR	LOCATION ELEC-TRONICS	PRESS. SHELL ELECT. PENE-TRATIONS	REMARKS
ACCELEROMETER PRIMARY AXIAL	1 ANALOG CHANNEL (0-5VDC)	FROM 4 X 10^{-4} G (141 KM) TO SURFACE	10	1/1	8/1	8/1	2.5/1	1/1	1/20	6/1	INSIDE PRESSURE SHELL AT CENTER OF GRAVITY	SAME	5	CALIBRATION BY TURN-ON AND TRANSMISSION FOR FEW MINUTES PRIOR TO RELEASE FROM BUS. FIVE UMBILICAL WIRES REQUIRED FOR PRELAUNCH CHECKOUT. NO PENETRATIONS OTHERWISE.
BACKUP AXIAL	1 ANALOG CHANNEL (0-5VDC)		10	1/1	-	-	2.5/1	1/1	1/20	-				
Y-LATERAL	1 ANALOG CHANNEL (0-5VDC)		10	1/1	-	-	2.5/1	1/1	1/40	-				
Z-LATERAL	1 ANALOG CHANNEL (0-5VDC)		10	1/1	-	-	2.5/1	1/1	1/40	-				
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7	1/140	-	1/140	1/140	1/140	1/140	1/140				
TURBULENCE	1 ANALOG CHANNEL (0-5VDC)		7	-	-	-	-	1/7	1/10	-				
TEMPERATURE ATMOSPHERE TEMPERATURE	1 ANALOG CHANNEL (0-5VDC)	FROM 70 KM TO SURFACE	10	SEE REMARKS (1)	-	-	-	-	1/8 (2)	-	OUTSIDE DESCENT CAPSULE	INSIDE PRESS. SHELL	4	(1) CALIBRATION BY TURN-ON AND TRANSMISSION FOR FEW MINUTES PRIOR TO RELEASE FROM BUS. (2) 66 KM TO 42.9 KM (3) 42.9 KM TO SURFACE
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	1/7 (3)	-				
PRESSURE PRESSURE	1 ANALOG CHANNEL (0-5VDC)	FROM 70 KM TO SURFACE	10	SEE REMARKS (1)	-	-	-	-	1/8 (2)	-	INSIDE PRESSURE SHELL	SAME	0	(1) CALIBRATION BY TURN-ON AND TRANSMISSION FOR FEW MINUTES PRIOR TO RELEASE FROM BUS. (2) 66 KM TO 42.9 KM (3) 42.9 KM TO SURFACE
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	1/7 (3)	-				
CLOUD PARTICLE SIZE ANALYZER SCIENCE AND HOUSEKEEPING	1 DIGITAL SERIAL	FROM 70 KM TO SURFACE	240 (1)		-	-	-	-	1/8 (2) 1/7 (3)	-	INSIDE PRESSURE SHELL	SAME	0	(1) 30 WORDS AT 8 BITS EACH. WORD AND BIT CLOCK SYNC SIGNALS REQUIRED. (2) 66 KM TO 42.9 KM (3) 42.9 KM TO SURFACE
SOLAR FLUX RADIOMETER SCIENCE AND HOUSEKEEPING	1 DIGITAL SERIAL	FROM 70 KM TO SURFACE	240 (1) 72 (2)	NONE DEFINED	-	-	-	-	1/30 (1) 1/26 (2)	- -	INSIDE PRESSURE SHELL	SAME	0	TIMING PULSE (1 S) REQUIRED; DATA READ-OUT SYNC PULSE REQUIRED. (1) 66 KM TO 42.9 KM (2) 42.9 KM TO SURFACE
HYGROMETER HUMIDITY	1 ANALOG CHANNEL (0-5VDC)	FROM 70 KM TO 42.9 KM (1)	10	NONE DEFINED	-	-	-	-	1/20	-	OUTSIDE DESCENT CAPSULE	INSIDE PRESS. SHELL	8	(1) TURN-OFF AT 42.9 KM
RANGE	1 ANALOG CHANNEL (0-5VDC)		1		-	-	-	-	1/20	-				
HOUSEKEEPING	1 ANALOG CHANNEL (0-5VDC)		10		-	-	-	-	1/20	-				
GAS CHROMATO-GRAPH	1 DIGITAL SERIAL	FROM 70 KM TO SURFACE	11		-	-	-	-	-	-	INSIDE PRESSURE SHELL	SAME	0	
	2 ANALOG CHANNEL (0-5VDC)		-		-	-	-	-	2/1	-				
WIND-ALTITUDE RADAR SCIENCE	1 DIGITAL SERIAL	40 KM TO SURFACE	35		-	-	-	-	1/14		OUTSIDE PRESSURE VESSEL	INSIDE PRESS. SHELL	2	
VOLTAGE	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	-					
TEMPERATURE	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	-					
MASS SPECTROMETER	1 DIGITAL SERIAL	FROM 70 KM TO SURFACE	8000 (1)		-	-	-	-	1/88 (3) 1/140 (4)	-	INSIDE PRESSURE SHELL	SAME	0	(1) 10 BIT WORDS (2) WORD AND BIT RATE TIMING SIGNALS TBD. (3) 66 KM TO 42.9 KM (4) 42.9 KM TO SURFACE
PLANETARY FLUX RADIOMETER SCIENCE AND HOUSEKEEPING	1 DIGITAL SERIAL (1)	FROM 70 KM TO SURFACE	100	NONE DEFINED	-	-	-	-	1/30 (2) 1/26 (3)	-	INSIDE PRESSURE SHELL	SAME	0	(1) TIMING PULSE (1 S) (2) 70 KM TO 42.9 KM (3) 42.9 KM TO SURFACE

NOTE: HOUSEKEEPING REQUIREMENTS TBD.

Table 6B-2. Engineering Measurement List - Large Probe

MEASUREMENT DESCRIPTION	TYPE	RANGE	ACCURACY (2)	ENTRY (1)	DESCENT (1)
ELECTRICAL POWER AND PYROTECHNICS					
BATTERY TERMINAL VOLTAGE	2A	0/32 VDC	2.0	1/400	1/400
BATTERY INTERNAL TEMPERATURE	2A	0/200 °F	2.0		1/400
BATTERY CURRENT	3A	0/15 A	3.0	1/400	
POWER SWITCH MONITOR (SAFE, ARM)	2B	ON/OFF	N/A	1/200	
POWER SWITCH MONITOR (POWER TRANSFER)	B	ON/OFF	N/A		1/200
IR HEATER SWITCH	B	ON/OFF	N/A	1/200	1/200
WINDOW HEATER SWITCH	B	ON/OFF	N/A	1/200	1/200
BATTERY HEATER SWITCH	B	ON/OFF	N/A	1/200	1/200
TEMPERATURE PRESSURE EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
SOLAR FLUX AND PLANETARY FLUX EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
MASS SPECTROMETER EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
CLOUD PARTICLES SIZE EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
WIND ALTITUDE RADAR EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
ACCELEROMETER EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
HYGROMETER EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
GAS CHROMATOGRAPH EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
DATA HANDLING AND COMMAND SUBSYSTEM					
SEQUENCER STATUS - BIT 1	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 2	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 3	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 4	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 5	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 6	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 7	B	ON/OFF	N/A	1/200	1/200
SEQUENCER STATUS - BIT 8	B	ON/OFF	N/A	1/200	1/200
REGULATED VOLTAGE +5 VDC	A	4.7/5.3 VDC	2.0		1/400
REGULATED VOLTAGE +12 VDC	A	11/13 VDC	2.0		1/400
REGULATED VOLTAGE -12 VDC	A	-13/-11 VDC	2.0		1/400
REGULATED VOLTAGE -16 VDC	A	-17/-15 VDC	2.0		1/400
A/D CALIBRATION VOLTAGE (LOW)	A	0/250 mV	0.2		1/800
A/D CALIBRATION VOLTAGE (MED)	A	2.2/2.6 V	0.2		1/800
A/D CALIBRATION VOLTAGE (HIGH)	A	4.5/5.5 V	0.2		1/800
SCID. EXTENDED FRAME COUNTER	6B	ON/OFF	N/A		1/400
COMMUNICATIONS SUBSYSTEM					
POWER AMPLIFIER OUTPUT	2A	0/24 W	2.0	1/400	1/400
POWER AMPLIFIER TEMPERATURE	2A	-28/200 °F	5.0	1/400	1/400
POWER AMPLIFIER INTERNAL TEMPERATURE	2A	-28/200 °F	3.0	1/400	1/400
CURRENT POWER AMPLIFIER INPUT	2A	0/5 A	5.0	1/400	1/400
DRIVER POWER OUTPUT	A	0 TO 1.5 W	2.0	1/400	1/400
AUXILLARY OSCILLATOR TEMPERATURE	A	-25 TO +200°F	3.0	1/400	1/400
TEMPERATURE, DRIVER OUTPUT STAGE	A	-25 TO +200°F	5.0	1/400	1/400
RECEIVER MODE INDICATION	B	SEARCH/LOCK	N/A	1/200	1/200
RECEIVER STATIC PHASE ERROR	A	-30/+30°	4.0	1/400	1/400
RECEIVER AGC	A	-148/80 DBM	2.0	1/400	1/400
VCO TEMPERATURE	A	-25 TO +200°F	3.0	1/400	1/400
THERMAL CONTROL, HEAT SHIELD SUBSYSTEM					
TEMPERATURE, AEROSHELL FOREBODY HEAT SHIELD BACKFACE	A	-150/+600 °F	±15 °F	1/400	
TEMPERATURE, AEROSHELL AFTERBODY HEAT SHIELD BACKFACE	A	-150/+600 °F	±15 °F	1/400	
PRESSURE, PROBE INTERIOR	A	0/25 PSIA	1.0 PSIA		1/400
TEMPERATURE, EQUIPMENT PLATFORM (1)	A	0/150 °F	±3 °F		1/400
TEMPERATURE, EQUIPMENT PLATFORM (2)	A	0/150 °F	±3 °F		1/400
TEMPERATURE, INSULATION EXTERIOR (1)	A	-60/1000 °F	±25 °F		1/400
TEMPERATURE, INSULATION EXTERIOR (2)	A	-60/1000 °F	±25 °F		1/400
TEMPERATURE, INSULATION EXTERIOR (3)	A	-60/1000 °F	±25 °F		1/400
TEMPERATURE, PRESSURE SHELL (INTERIOR) (1)	A	0/300 °F	±6 °F		1/400
TEMPERATURE, PRESSURE SHELL (INTERIOR) (2)	A	0/300 °F	±6 °F		1/400
TEMPERATURE, PRESSURE SHELL (INTERIOR) (3)	A	0/300 °F	±6 °F		1/400
TEMPERATURE, IRFR WINDOW LENS	A	-60/1000 °F	±25 °F		1/400
TEMPERATURE, IRFR WINDOW TUBE	A	-60/1000 °F	±25 °F		1/400

NOTES:

(1) COLUMN ENTRY IS MINIMUM SAMPLE RATE IN SAMPLES/SECOND. MORE SAMPLES ARE ACCEPTABLE. ENTRY PHASE ENDS AT APPROXIMATELY 6120 KM. DESCENT PHASE IS 6120 TO 6050 KM.

(2) THE ACCURACY SPECIFIED IS APPLICABLE TO THE PROBE ONLY (FROM THE ENVIRONMENT OR PARAMETER BEING MONITORED TO THE A/D CONVERTER OUTPUT). GROUND DECODER AND PROCESSING ERROR CONTRIBUTIONS ARE NOT INCLUDED. IN THOSE ENTRIES WHERE ENGINEERING UNITS ARE NOT INCLUDED, THE VALUE IS GIVEN IN ± PERCENT OF FULL SCALE.

ABBREVIATIONS: A - ANALOG, B - BILEVEL, D - DIGITAL

Table 6B-3. Science Instrument Telemetry Signal Characteristics - Small Probe

INSTRUMENT	SIGNAL CHARACTERISTICS	OPERATING ALTITUDE	BITS/ SAMPLE	SAMPLES/SECOND							LOCATION		PRESS. SHELL ELECT. PENE- TRATIONS	REMARKS
				CALI- BRATION	PRE- ENTRY	ENTRY	BLACK- OUT	POST- BLACK- OUT	TERMINAL DESCENT	POST- IMPACT	SENSOR	ELEC- TRONICS		
ACCELEROMETER		FROM 4 X 10^{-4} G (141 KM) TO SURFACE									INSIDE PRESSURE SHELL AT CENTER OF GRAVITY	SAME	5	
AXIAL ACCELER- OMETER	1 ANALOG CHANNEL (0-5VDC)		10	1/1	1/1	1/1	1/1	1/1	1/20	1/1				
TURBULENCE	1 ANALOG CHANNEL (0-5VDC)		7	-	-	-	-	-	1.3/1	-				
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7	1/140	-	1/140	1/140	1/140	1/140	1/140				
TEMPERATURE		FROM 70 KM TO SURFACE		S/S TBD SEE REMARKS							OUTSIDE PRESSURE SHELL	INSIDE PRESS. SHELL	4	CALIBRATION BY TURN-ON PRIOR TO RELEASE FROM BUS
TEMPERATURE	1 ANALOG CHANNEL (0-5VDC)		10		-	-	-	-	0.6/1	-				
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	1/140	-				
PRESSURE		FROM 70 KM TO SURFACE		S S TBD (SEE REMARKS)							INSIDE PRESSURE SHELL	SAME	0	CALIBRATION BY TURN-ON PRIOR TO RELEASE FROM BUS
PRESSURE	1 ANALOG CHANNEL (0-5VDC)		10		-	-	-	-	0.6/1	-				
THERMISTOR	1 ANALOG CHANNEL (0-5VDC)		7		-	-	-	-	1/140	-				
NEPHELOMETER		FROM 70 KM TO SURFACE		TBD							INSIDE PRESSURE SHELL	SAME	0	(1) REQUIRES TIMING SIGNAL, CHARAC-TERISTICS TBD (2) SERIAL FORM IN-TERPRETED
SCIENCE	1 DIGITAL SERIAL		43		-	-	-	-	0.6/1	-				
CALIBRATION	1 DIGITAL SERIAL		10		-	-	-	-	1/900	-				
IR FLUX DETECTOR	1 ANALOG CHANNEL (0-5VDC)	FROM 70 KM TO SURFACE	8						1/30	-	INSIDE PRESSURE SHELL	SAME	0	
	1 ANALOG CHANNEL (0-5VDC)		8		-	-	-	-	1/60	-				
	1 ANALOG CHANNEL (0-5VDC)		8						1/60					

NOTES: (1) REQUIREMENTS ARE BASED UPON SCIENCE VERSION IV, DATED 13 .
(2) HOUSEKEEPING REQUIREMENTS TBD EXCEPT FOR MAGNETOMETER

Table 6B-4. Engineering Measurement List - Small Probe

MEASUREMENT DESCRIPTION	TYPE	RANGE	ACCURACY (3)	ENTRY (2)	DESCENT (2)
BATTERY TERMINAL VOLTAGE	A	0/32 VDC	2.0	1/600	1/600
BATTERY TEMPERATURE	A	0/200 °F	2.0		1/600
BATTERY CURRENT	A	0/10 A	3.0	1/600	
POWER SWITCH MONITOR (POWER TRANSFER)	B	ON/OFF	N/A	1/600	
POWER SWITCH MONITOR (SAFE/ARM)	B	ON/OFF	N/A		1/600
WINDOW HEATER POWER	B	ON/OFF	N/A	1/200	1/200
S-BAND AMPLIFIER POWER	B	ON/OFF	N/A	1/200	1/200
TEMPERATURE EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
ACCELEROMETER EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
PRESSURE EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
STABLE OSCILLATOR POWER	B	ON/OFF	N/A	1/200	1/200
NEPHELOMETER EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
TRANSMITTER DRIVER POWER	B	ON/OFF	N/A	1/200	1/200
IR FLUX DETECTOR EXPERIMENT POWER	B	ON/OFF	N/A	1/200	1/200
DATA HANDLING AND COMMAND SUBSYSTEM					
SEQUENCER STATUS - BIT 1	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 2	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 3	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 4	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 5	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 6	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 7	B	ON/OFF	N/A	1/600	1/600
SEQUENCER STATUS - BIT 8	B	ON/OFF	N/A	1/600	1/600
REGULATED VOLTAGE +5 VDC	A	4.7/5.3 VDC	±0.2 VDC		1/600
REGULATED VOLTAGE +12 VDC	A	11/13 VDC	±0.2 VDC		1/600
REGULATED VOLTAGE -12 VDC	A	-13/-11 VDC	±0.2 VDC		1/600
REGULATED VOLTAGE -16 VDC	A	-17/-15 VDC	±0.2 VDC		1/400
A/D CALIBRATION VOLTAGE (LOW)	A	0/250 mV	±0.2 VDC		1/800
A/D CALIBRATION VOLTAGE (MEDIUM)	A	2.2/2.6 V	±0.2 VDC		1/800
A/D CALIBRATION VOLTAGE (HIGH)	A	4.5/5.5 V	±0.2 VDC		1/800
SCID EXTENDED FRAME COUNTER	6B	ON/OFF	N/A		1/400
COMMUNICATIONS SUBSYSTEM					
POWER AMPLIFIER OUTPUT	A	0/24 W	2.0	1/600	1/600
POWER AMPLIFIER TEMPERATURE	A	-28/200 °F	3.0	1/600	1/600
POWER AMPLIFIER INPUT CURRENT	A	0/5 A	5.0	1/600	1/600
TRANSMITTER DRIVER REFERENCE OSCILLATOR TEMPERATURE	A	-25/200 °F	3.0	1/600	1/600
STABLE OSCILLATOR TEMPERATURE	2A	-25/200 °F	3.0	1/600	1/600
OUTPUT LEVEL	A	0/1.5 W	2.0	1/600	1/600
THERMAL CONTROL/HEAT SHIELD SUBSYSTEM					
TEMPERATURE, FOREBODY H/S BACKFACE	A	-150/900 °F	±25 °F	1/600	
PRESSURE, PROBE INTERIOR	A	0/25 PSIA	1.0 PSIA		1/600
TEMPERATURE, EQUIPMENT PLATFORM (1)	A	0/150 °F	±3 °F	1/600	
TEMPERATURE, EQUIPMENT PLATFORM (2)	A	0/150 °F	±3 °F	1/600	
TEMPERATURE, INSULATION EXTERIOR (1)	A	-60/1000 °F	±25 °F		1/600
TEMPERATURE, INSULATION EXTERIOR (2)	A	-60/1000 °F	±25 °F		1/600
TEMPERATURE, INSULATION EXTERIOR (3)	A	-60/1000 °F	±25 °F		1/600
TEMPERATURE, PRESSURE SHELL INTERIOR (1)	A	0/300 °F	±6 °F		1/600
TEMPERATURE, PRESSURE SHELL INTERIOR (2)	A	0/300 °F	±6 °F		1/600
TEMPERATURE, PRESSURE SHELL INTERIOR (3)	A	0/300 °F	±6 °F		1/600
TEMPERATURE, NEPHELOMETER WINDOW LENS	A	-60/1000 °F	±25 °F		1/400
TEMPERATURE, NEPHELOMETER WINDOW TUBE	A	-60/1000 °F	±25 °F		1/400

NOTES:

(1) THIS LIST IS FOR ONE PROBE; DATA REQUIREMENTS FOR THE THREE PROBES ARE IDENTICAL.

(2) COLUMN ENTRY IS MINIMUM SAMPLE RATE IN SAMPLES/SECOND. MORE SAMPLES ARE ACCEPTABLE. ENTRY PHASE ENDS AT APPROXIMATELY 6120 KM. DESCENT PHASE IS 6120 TO 6050 KM.

(3) THE ACCURACY SPECIFIED IS APPLICABLE TO THE PROBE ONLY (FROM THE ENVIROMENT OR PARAMETER BEING MONITORED TO THE A/D CONVERTER OUTPUT). GROUND DECODER AND PROCESSING ERROR CONTRIBUTIONS ARE NOT INCLUDED. IN THOSE ENTRIES WHERE ENGINEERING UNITS ARE NOT INCLUDED, THE VALUE IS GIVEN IN ± PERCENT OF FULL SCALE.

ABBREVIATIONS: A ANALOG, B BILEVEL, D DIGITAL

APPENDIX 6C

MISSION PROFILE SUMMARY OF MAJOR EVENTS (TYPICAL)

APPENDIX 6C

MISSION PROFILE SUMMARY OF MAJOR EVENTS (TYPICAL)

Event	Reference Time	Remarks
PRELAUNCH (F-11 TO F-0 DAYS)		
Spacecraft weighing and mating to third stage	F-15 days	
Spacecraft - third stage mated to launch vehicle	F-12	
Vehicle all systems test	F-5	
Spacecraft final composite readiness test	F-0	F-0 is launch day.
LIFTOFF TO CRUISE		Refer to Table 6C-2 for detailed sequence.
Liftoff	L-0	
Precess to cruise attitude	L+1 hr	
CRUISE TO PRESEPARATION CHECKOUT		Refer to Table 6C-3 for detailed sequence.
First midcourse maneuver	L+5 days	
Second midcourse maneuver	L+15 days	
Third midcourse maneuver	E-30 days	E is entry time of large probe
PRESEPARATION CHECKOUT (4 PROBES)	E-29 days	Refer to Table 6C-4 for detailed sequence.

NOTE: Probes are released every four days, starting with large probe release at E-25 days. The following overview gives the large probe major sequences from preseparation to impact, and then summarizes the small probe sequences for the corresponding period.

LARGE PROBE

Event	Reference Time	Remarks
PRESEPARATION CALIBRATION AND SEPARATION		Refer to Table 6C-5 for detailed sequence.
Preseparation Power On	S-5 min	
Preseparation Power Off	S-1 min	Sequence calibration data acquired.
Release Large Probe	S=0, E-25 days	
BUS RETARGET MANEUVER	E-23 days	Reference; not probe sequence.
SEPARATION TO POST IMPACT		Refer to Table 6C-6 for detailed sequence.
IR Reference Heater On	E-2 days	
Science On	E-10 min	Power On.
Entry Point	E=0	Reference only. H = 250 km.
Sense 50-g Increasing	T=0	
Mortar Fire	T+21 s	Deploys parachute
Forebody Aeroshell Release	T+26 s	
Afterbody/Parachute Release	T+39 min, 42 s	Releases descent capsule.
Impact	T+73 min, 12s	

SMALL PROBES

Event	Reference Time	Remarks
PRESEPARATION CALIBRATION AND SEPARATION		Refer to Table 6C-7 for detailed sequences.
Preseparation Power On	S_i-5 min	Typical for all Small Probes.
Preseparation Power Off	S_i-4 min	
Release Small Probe-1	S_i=0, E-21 days	
(Same sequence for SP-2, SP-3)		
Bus Retarget Maneuver	E-19 days	Reference; not probe sequence.
Release Small Probe-2	E-17 days	
Bus Retarget Maneuver	E-15 days	Reference; not probe sequence.
Release Small Probe-3	E-13 days	
Bus Retarget Maneuver	E-11 days	Reference; not probe sequence.
SEPARATION TO POST IMPACT		Refer to Table 6C-8 for detailed sequence.
Stable Oscillator-Power On	E_i-60 min	E_i = entry time for 3 small probes.
Entry	E_i=0	Reference only. H = 250 km.
50-g Increasing	T_i=0	Typical all Small Probes.
Impact	T_i+65 min	Typical all Small Probes.

Table 6C-1. Prelaunch (F-11 to F-0 Days)

EVENT NO.	EVENT	DATE/TIME	REMARKS
			PRELAUNCH SEQUENCE IS PRESENTED FOR INFORMATION ONLY; APPROXIMATE TIMES.
101	SPACECRAFT TO SPIN FACILITY	F-11 DAYS	HAZARDOUS SYSTEMS PREPARATION. MAY BE SCHEDULED EARLIER.
102	SPACECRAFT WEIGHING AND MATING TO THIRD STAGE.	F-10	
103	SPIN BALANCE SPACECRAFT - THIRD STAGE	F-9	
104	SPACECRAFT - THIRD STAGE IN TRANSPORT CANISTER	F-8	
105	SPACECRAFT - THIRD STAGE MATED TO LAUNCH VEHICLE	F-7	
106	SPACECRAFT INTEGRATED SYSTEMS TEST	F-6	VERIFY SPACECRAFT AND EXPERIMENTS. SAME AS AN INFLIGHT PRESEPARATION CHECKOUT.
107	VEHICLE ALL SYSTEMS TEST	F-5	COMPLETE ELECTRICAL CHECK FOLLOWED BY SIMULATED LAUNCH. SPACECRAFT RF AND COMMAND SYSTEMS RADIATING, CRITICAL SWITCHING FUNCTIONS EXERCISES, DATA MONITORED FOR EMI ANALYSIS.
108	LAUNCH COMPLEX READINESS TEST	F-4	
109	VEHICLE ORDNANCE INSTALLATION	F-3	
110	VEHICLE - SPACECRAFT INTEGRATED SYSTEMS TEST. FINAL ASSEMBLY OF SPACECRAFT CONFIGURATION STARTED.	F-2	
111	SPACECRAFT ELECTRICAL CHECK, ORDNANCE INSTALLATION, AND IN FINAL PHYSICAL CONFIGURATION PRIOR TO FAIRING INSTALLATION.	F-1	SPACECRAFT ELECTRICAL CHECK SAME AS AN INFLIGHT CRUISE CHECKOUT
112	LAUNCH DAY. SPACECRAFT FINAL COMPOSITE READINESS TEST.	F-0	450 TO 295 MIN PRIOR TO LIFTOFF.
113	ORDNANCE HOOKUP	F-0	325 TO 175 MIN PRIOR TO LIFTOFF.
114	TOWER REMOVAL	F-0	175 TO 55 MIN PRIOR TO LIFTOFF.

Table 6C-2. Liftoff to Cruise

EVENT NO.	EVENT	TIME	REMARKS
			APPROXIMATE SEQUENCE; FOR INFORMATION ONLY
201	LIFTOFF	L-0	TWO-INCH MOTION
202	ROLL PROGRAM INITIATION	L+2 S	
203	BOOSTER ENGINE CUTOFF	L+153 S	
204	BOOSTER PACKAGE JETTISON	L+156 S	
205	JETTISON INSULATION PANELS	L+201 S	
206	SUSTAINER ENGINE CUTOFF	L+251 S	PROPELLANT DEPLETION
207	ATLAS/CENTAUR SEPARATION	L+253 S	
208	MES 1	L+263 S	
209	JETTISON NOSE FAIRING	L+275 S	
210	MECO 1	L+586 S	PARKING ORBIT
211	MECO 2	L+25 MIN	
212	PRECESS TO CRUISE ATTITUDE	L+1 HR	

Table 6C-3. Cruise to Preseparation Checkout

EVENT NO.	EVENT	REFERENCE TIME	REMARKS
			NO LOW-LEVEL PROBE TELEMETRY IS ACQUIRED IN THIS MISSION PHASE. CAPABILITY SHELL EXIST TO ACQUIRE PROBE DATA USING PRESEPARATION CHECKOUT SEQUENCE BY GROUND COMMAND
240	FIRST MIDCOURSE MANEUVER TRIM ATTITUDE PRECESSION ΔV AND DELAYS PRECESSION	L+5 DAYS	
241	SECOND MIDCOURSE MANEUVER TRIM ATTITUDE PRECESS ΔV TRIM ΔV RETURN PRECESSION	L+15 DAYS	
242	THIRD MIDCOURSE MANEUVER TRIM ATTITUDE TWO HOUR PREPARATION PRECESS ΔV AND DELAY TRIM ΔV RETURN PRECESSION	E-30 DAYS	E = 0 IS ENTRY TIME OF LARGE PROBE

Table 6C-4. Preseparation Checkout (4 Probes)

EVENT NO.	EVENT	TIME	REMARKS
250	LOAD COMMAND WORDS FOR LARGE PROBE C/O INTO BUS STORAGE	C_o - 220 S (MINIMUM)	DSN UPLINK; CHECKOUT STARTS 29 DAYS PRIOR TO ENTRY.
251	APPLY BUS POWER TO LARGE PROBE	C_o = 0	DSN UPLINK
	DTU ON (FORMAT A) TRANSDUCERS ON START DESCENT TIMER SCIENCE INSTRUMENTS OFF	C_o = 0	PROBE PCU INTERNAL
252	BEGIN LOADING INITIALIZATION COMMANDS INTO PCU	C_o+2 SEC	FROM BUS
	ACCELERATE DTU BIT RATE TO 512 BPS		
	INHIBIT PYRO ARM INHIBIT MASS SPECTROMETER HEATER INHIBIT WINDOW HEATERS INHIBIT TRANSMITTER AND RECEIVER ON INHIBIT CLOUD PARTICLE SIZE ANALYZER INHIBIT WIND ALTITUDE RADAR SCIENCE POWER ON		LIMITS POWER REQUIRED FROM BUS
	ACTIVATE PCU INITIALIZATION COMMANDS	C_o+18 S	PROBE PCU INTERNAL (UPON SENSING 16TH COMMAND BIT)
253	APPLY SIMULATED 50-G SIGNAL	C_o+34 S	FROM BUS; SEQUENCE FROM E-10 MIN IS CHECKED OUT USING ACCELERATED DESCENT SEQUENCE.
254	MASS SPECTROMETER OFF; CLOUD PARTICLE ANALYZER ON	C_o+2 MIN 40.6 S	DSN UPLINK
254.1	ALL SCIENCE OFF WIND DRIFT RADAR ON 40.0		
254.2	ALL SCIENCE OFF		
255	POWER TRANSFER TO PROBE INTERNAL BATTERY	C_o+7 MIN 4 S	DSN UPLINK
256	DISCONNECT BUS POWER	C_o+7 MIN 6 S	FROM BUS
257	PROBE TRANSMITTER AND RECEIVER ON	C_o+7 MIN 8 S	FROM BUS
258	PROBE TRANSMITTER OFF	C_o+8 MIN 8 S	FROM BUS
	PROBE OFF	C_o+8 MIN 8 S	
259	PROBE TRANSMITTER AND PROBE OFF BACKUP	C_o+8 MIN 20 S	DSN UPLINK

Table 6C-4. Preseparation Checkout (4 Probes) (Continued)

EVENT NO.	EVENT	TIME	REMARKS
260	LOAD COMMAND WORDS FOR SMALL PROBE 1 C/O INTO BUS STORAGE	C_1-220 S (MINIMUM)	DSN UPLINK: $C_1 = C_0$+13 MIN
261	APPLY BUS POWER TO PROBE	C_1=0	DSN UPLINK
	DTU ON (FORMAT A) TRANSDUCERS ON START DESCENT TIMER SCIENCE INSTRUMENTS OFF		PROBE PCU INTERNAL
262	BEGIN LOADING INITIALIZATION COMMANDS INTO PCU:	C_1+2 S	FROM BUS
	ACCELERATE DTU BIT RATE TO 512 BPS INHIBIT PYRO ARM INHIBIT WINDOW HEATER INHIBIT TRANSMITTER ON SCIENCE POWER ON		
	ACTIVATE PCU INITIALIZATION COMMANDS	C_1+18 S	PROBE PCU INTERNAL (UPON SENSING 16TH COMMAND BIT)
263	APPLY SIMULATED 50-G SIGNAL	C_1 + 34 S	FROM BUS; SEQUENCE FROM E-5 MIN IS CHECKED OUT USING ACCELERATED DESCENT SEQUENCE.
264	POWER TRANSFER TO PROBE INTERNAL BATTERY	C_1+2 MIN	DSN UPLINK
265	DISCONNECT BUS POWER	C_1+2 MIN 2 S	FROM BUS
266	PROBE TRANSMITTER ON	C_1+2 MIN 16 S	FROM BUS
267	PROBE TRANSMITTER OFF PROBE OFF	C_1+3 MIN 16 S	FROM BUS
268	PROBE TRANSMITTER AND PROBE OFF - BACKUP	C_1+3 MIN 30 S	DSN UPLINK
	SMALL PROBES -2 AND -3 ARE CHECKED OUT IN SAME SEQUENCE AS FOR SMALL PROBE -1. CHECKOUT ON SMALL PROBE -3 ENDS AT APPROXIMATELY C_0 + 33 MINUTES.		

Table 6C-5. Preseparation Calibration and Separation - Large Probe

EVENT NO.	EVENT	TIME (HOURS:MINUTES:SECONDS)	REMARKS
270	PRESEPARATION POWER ON	S - 00:05:00	BUS POWER
271	POWER ON: DTU ON (FORMAT A) ACCELEROMETER ON TEMPERATURE SENSING SYSTEM ON PRESSURE SENSING SYSTEM ON	S - 00:05:00	
272	DTU TO FORMAT D_1	S - 00:04:00	3 MINUTES OF SCIENCE CALIBRATION DATA
273	POWER OFF: PRESSURE SENSING SYSTEM OFF TEMPERATURE SENSING SYSTEM OFF ACCELEROMETER OFF	S - 00:01:00	
274	DTU POWER OFF	S - 00:01:00	
275	PRESEPARATION POWER OFF	S - 00:01:00	
276	DISCONNECT LARGE PROBE UMBILICAL	S - 00:00:15	
301	RELEASE LARGE PROBE	S = 0	LARGE PROBE RELEASED AT ENTRY MINUS 25 DAYS.

Table 6C-6. Separation to Post Impact: Large Probe

EVENT NO.	EVENT	TIME (HOURS:MINUTES:SECONDS)	REMARKS
301	SEPARATION	S = 0	ENTRY MINUS 25 DAYS
302	IR REFERENCE HEATER ON	E - 2 DAYS	TIMED EVENT WITH 25 - DAY COAST TIMER. HEATER ON FOR DURATION OF MISSION
303	BATTERY HEATER ON	E - 2:45:00	
304	POWER ON ENGINEERING SUBSYSTEMS: TRANSMITTER DRIVER ON POWER AMPLIFIER ON RECEIVER ON DTU ON (FORMAT A) TRANSDUCERS ON INITIALIZE DESCENT TIMER BATTERY HEATER OFF	E - 00:45:00	TIMED EVENT BY 25 - DAY COAST TIMER FOR DSN ACQUISITION.
305	POWER OFF ENGINEERING SUBSYSTEMS: TRANSMITTER DRIVER OFF POWER AMPLIFIER OFF RECEIVER OFF DTU OFF TRANSDUCERS OFF	E - 00:35:00	
306	SCIENCE POWER ON DTU ON (FORMAT A) TRANSDUCERS ON	E - 00:10:00	TIMED EVENT BY DESCENT TIMER. ACQUIRE AND STORE ACCELEROMETER AND ENGINEERING DATA AT 128 BPS.
400	ENTRY POINT	E = 0	FOR REFERENCE ONLY H - 250 KM ABOVE MSL
401	4 X 10^{-4} G-POINT	E + 00:00:17	FOR REFERENCE ONLY H = 140 KM ABOVE MSL.
402	START BLACK-OUT (APPROXIMATE)	E + 00:00:21	REFERENCE ONLY
403	SENSE 50-G INCREASING	E + 00:00:25, T = 0	H = 92 KM ABOVE MSL
404	RESET DESCENT TIMER	T = 0	TRIGGERED BY REDUNDANT G-SWITCHES
405	BACKUP POWER ON	T = 0	TRIGGERED BY REDUNDANT G-SWITCHES
406	DTU TO FORMAT B	T = 0	DATA STORED
407	WINDOW HEATER - POWER ON	T = 0	
408	PEAK G'S	T + 00:00:02	REFERENCE ONLY. H = 81 KM ABOVE MSL.
409	DATA ACQUISITION RATE SWITCHED TO 64 BPS PYROS ARM	T + 00:00:06	END BLACKOUT (APPROXIMATE)
410	TRANSMITTER DRIVER ON POWER AMPLIFIER ON RECEIVER ON	T + 00:00:10	TRANSMITTER WARMUP - CONTINUE DATA STORAGE
411	MORTAR FIRE	T + 00:00:21	DEPLOYS PARACHUTE. TIMED EVENT TO OCCUR AT M - 0.78, Q = 35 PSF.
500	AEROSHELL FOREBODY JETTISON	T + 00:00:26	RELEASES AEROSHELL. ALLOWS 5 SECONDS FOR STABILIZATION. H = 70 KM.
501	DTU TO FORMAT D_1, DATA ACQUISITION RATE 128 BPS	T + 00:00:26	ALL SCIENCE DATA ACQUIRED AND TRANSMITTED R/T.
502	MASS SPECTROMETER INLET CAP EJECTION	T + 00:00:31	PYRO EVENT.
503	NEUTRAL MASS SPECTROMETER HEATER ON-LOW	T + 00:00:31	
504	MASS SPECTROMETER - OPEN TUBE 1	T + 00:01:01	30 S AFTER INLET CAP EJECTION
505	FIRST GC SAMPLE INJECT	T + 00:02:46	
506	MASS SPECTROMETER - CLOSE TUBE 1	T + 00:12:21	11:20 MIN BETWEEN OPEN AND CLOS-
507	FIRST GC SAMPLE CLOSE	T + 00:12:46	
508	MASS SPECTROMETER - OPEN TUBE 2	T + 00:13:01	12 MIN BETWEEN TUBE OPENINGS
509	SECOND GC SAMPLE INJECT	T + 00:22:46	
510	MASS SPECTROMETER - CLOSE TUBE 2	T + 00:24:21	11:20 MIN BETWEEN OPEN AND CLOSE
511	MASS SPECTROMETER - OPEN TUBE 3	T + 00:25:01	12 MIN BETWEEN TUBE OPENINGS
512	SECOND GC SAMPLE CLOSE	T + 00:32:46	
513	MASS SPECTROMETER - CLOSE TUBE 3	T + 00:36:21	11:20 MIN BETWEEN OPEN AND CLOSE
514	MASS SPECTROMETER - OPEN TUBE 4	T + 00:37:01	12 MIN BETWEEN TUBE OPENINGS
515	AFTERBODY/PARACHUTE RELEASE	T + 00:39:42	H = 42.9 KM
516	NEUTRAL MASS SPECTROMETER HEATER ON-HIGH	T + 00:39:42	
517	DTU TO FORMAT D_2	T + 00:39:42	STORED AND R/T DATA TRANSMITTED. TIMED EVENT OCCURS AT H = 42.9 KM.
518	WIND ALTITUDE RADAR ON	T + 00:39:42	
519	HYGROMETER POWER OFF	T + 00:39:42	
520	THIRD GC SAMPLE INJECT	T + 00:42:46	
521	MASS SPECTROMETER - CLOSE TUBE 4	T + 00:48:21	11:20 MIN BETWEEN OPEN AND CLOSE
522	MASS SPECTROMETER - OPEN TUBE 5	T + 00:49:01	12 MIN BETWEEN TUBE OPENINGS
523	THIRD GC SAMPLE CLOSE	T + 00:52:46	
524	MASS SPECTROMETER - CLOSE TUBE 5	T + 00:60:21	11:20 MIN BETWEEN OPEN AND CLOSE
525	MASS SPECTROMETER - OPEN TUBE 6	T + 00:61:01	12 MIN BETWEEN TUBE OPENINGS
526	IMPACT	T + 00:73:12	

Table 6C-7. Preseparation Calibration and Separation - Small Probe(s)

EVENT NO.	EVENT	TIME (HOURS:MINUTES:SECONDS)	REMARKS
650	PRESEPARATION POWER ON	S_i - 00:05:00	BUS POWER; S_i = 0 IS SP-1 RELEASE TIME. IDENTICAL SEQUENCES USED FOR SP-2, SP-3.
651	POWER ON: ALL SCIENCE ON DTU ON (FORMAT A)	S_i - 00:05:00	
652	DTU TO FORMAT B	S_i - 00:04:00	3 MINUTES OF SCIENCE CALIBRATION DATA.
653	POWER OFF: ALL SCIENCE OFF (CALIBRATION DATA COMPLETED)	S_i - 00:01:00	
654	DTU POWER OFF	S_i - 00:01:00	
655	PRESEPARATION POWER OFF	S_i - 00:01:00	
656	DISCONNECT SMALL PROBE UMBILICAL	S_i - 00:00:15	

Table 6C-8. Separation to Post Impact: Small Probes

EVENT NO.	EVENT	TIME (HOURS:MINUTES:SECONDS)	REMARKS
801	SEPARATION	S = 0	SMALL PROBES RELEASED AT E - 21 DAYS, E - 17 DAYS, E - 13 DAYS. E IS ENTRY TIME (H = 250 KM ABOVE MSL)
802	BATTERY HEATER ON	E_1-4:40:00 (SP1) E_2-5:00:00 (SP2) E_2-4:40:00 (SP3)	
803	STABLE OSCILLATOR-POWER ON POWER ON ENGINEERING SUBSYSTEMS: TRANSMITTER DRIVER ON POWER AMPLIFIER ON DTU ON, FORMAT A TRANSDUCERS, ON INITIALIZE DESCENT TIMER BATTERY HEATER OFF	E_1-2:40:00 (SP1) E_2-3:00:00 (SP2) E_2-2:40:00 (SP3)	WARM-UP AND STABILIZE OSCILLATOR ALL DATA TRANSMITTED. DSN ACQUISITION.
804	POWER OFF ENGINEERING SUBSYSTEMS: TRANSMITTER DRIVER OFF POWER AMPLIFIER OFF DTU OFF TRANSDUCERS OFF	E_1-2:30:00 (SP1) E_2-2:50:00 (SP2) E_2-2:30:00 (SP3)	
805	BATTERY HEATER ON	E_1-1:00:00 (SP1) E_2-1:20:00 (SP2) E_2-1:00:00 (SP3)	
806	DTU ON (FORMAT A) TRANSDUCERS ON ALL SCIENCE ON BATTERY HEATER OFF	E_1-00:10:00 SP1) E_2-00:30:00 (SP2) E_2-00:10:00 (SP3)	ACCELEROMETER DATA TO STORAGE.
900	ENTRY	E = 0	REFERENCE ONLY H = 250 KM ALTITUDE ABOVE MSL
901	4×10^{-4} G DECELERATION	E + 00:00:11 (60° γ_E) E + 00:00:24 (25° γ_E)	REFERENCE ONLY
902	START BLACK-OUT (APPROXIMATE)	E + 00:00:14 (60° γ_E) E + 00:00:31 (25° γ_E)	REFERENCE ONLY
903	SENSE 50 G INCREASING	T = 0 E + 00:00:16 (60° γ_E) E + 00:00:34 (25° γ_E)	
904	RESET DESCENT TIMER	T = 0	
905	BACK-UP POWER ON	T = 0	
906	PIN PULLER ARMING	T = 0	
907	WINDOW HEATER - POWER ON	T = 0	
908	PEAK G'S	T + 00:00:02 E + 00:00:18 (60° γ_E) T + 00:00:03 E + 00:00:37 (25° γ_E)	REFERENCE ONLY
909	END BLACKOUT (APPROXIMATE)	T + 00:00:05 E + 00:00:21 (60° γ_E) T + 00:00:07 E + 00:00:41 (25° γ_E)	REFERENCE ONLY
910	TRANSMITTER DRIVER ON POWER AMPLIFIER ON	T + 00:00:09	TRANSMITTER WARMUP, CONTINUE DATA STORAGE.
911	DEPLOY SCIENCE, SCIENCE COVERS DTU TO FORMAT B	T + 00:00:16	ALL DATA ACQUIRED AND TRANSMITTED REAL TIME. ALTITUDE: 71.4 KM (25° γ_E) 66 KM (60° γ_E)
912	IMPACT	T + 00:65:00	

APPENDIX 6D

DETAILED WEIGHT BREAKDOWN
OPTIONAL ATLAS/CENTAUR ORBITER,
VERSION IV SCIENCE PAYLOAD

APPENDIX 6D

DETAILED WEIGHT BREAKDOWN OPTIONAL ATLAS/CENTAUR ORBITER, VERSION IV SCIENCE PAYLOAD

Despun Reflector, Option 1

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
ELECTRICAL POWER		44.5	98.1
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	15.10	33.3
BATTERY (16-15 A-HR NI-CD CELLS)	1	13.06	28.8
POWER CONTROL UNIT	1	6.35	14.0
CTRF	1	7.67	16.9
INVERTER	1	2.31	5.1
COMMUNICATIONS		12.7	27.9
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
POWER AMPLIFIER	4	1.09	2.4
TRANSMITTER DRIVERS	2	1.09	2.4
HYBRIDS	5	0.23	0.5
DIPLEXERS	2	1.95	4.3
SWITCHES	5	1.36	3.0
FORWARD OMNI ANTENNA	1	0.14	0.3
AFT OMNI ANTENNA	1	0.23	0.5
FANSCAN ANTENNA	1	0.45	1.0
HIGH-GAIN ANTENNA ASSEMBLY	1	2.04	4.5
RF CABLE AND CONNECTORS	AS REQUIRED	1.36	3.0
ELECTRICAL DISTRIBUTION		15.8	34.8
COMMAND DISTRIBUTION UNIT	1	4.45	9.8
HARNESS AND CONNECTORS	AS REQUIRED	11.34	25.0
DATA HANDLING		18.4	40.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	14.52	32.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		12.8	28.3
CONTROL ELECTRONICS ASSEMBLY	1	2.31	5.1
SUN SENSOR ASSEMBLY	2	0.41	0.9
DESPIN CONTROL ASSEMBLY	2	3.63	8.0
DESPIN DRIVE ASSEMBLY	1	6.49	14.3
PROPULSION		7.9	17.4
THRUSTERS	8	2.18	4.8
PROPELLANT TANK	3	4.49	9.9
FILTER	1	0.23	0.5
PRESSURE TRANSDUCER	1	0.14	0.3
FILL AND DRAIN VALVE	1	0.09	0.2
LINE/HEATER AND MISCELLANEOUS	AS REQUIRED	0.77	1.7
ORBIT INSERTION MOTOR (OIM) (BURNOUT)	1	18.7	41.1

Despun Reflector, Option 1 (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
THERMAL CONTROL		15.0	33.0
INSULATION	AS REQUIRED	8.12	17.9
FORWARD CLOSURE	1	1.32	2.9
AFT CLOSURE	1	0.54	1.2
LOUVERS	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	1.36	3.0
STRUCTURE		72.0	158.8
CENTRAL CYLINDER ASSEMBLY		22.77	50.2
UPPER RING	1		(6.4)
CYLINDER	1		(6.6)
PLATFORM RING	2		(3.4)
LOWER FRUSTUM	1		(11.2)
SEPARATION RING	1		(14.6)
OIM MOUNTING RING	1		(6.5)
ATTACH HARDWARE	AS REQUIRED		(1.5)
PLATFORM/COMPARTMENT ASSEMBLY		27.03	59.6
UPPER STRUTS	15		(5.7)
PLATFORM STRUTS	9		(3.7)
VERTICALS	9		(4.0)
UPPER RING ASSEMBLY	1		(5.6)
PLATFORM STRUT FITTINGS	9		(1.8)
PLATFORM ASSEMBLY	1		(34.6)
BRACKET AND ATTACH HARDWARE	AS REQUIRED		(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY		5.81	12.8
UPPER RING	1		(3.4)
LOWER RING	1		(4.0)
STRUTS	18		(5.4)
ANTENNA SUPPORT ASSEMBLY	1	1.32	2.9
MAGNETOMETER BOOM ASSEMBLY	1	4.22	9.3
PROPULSION SUPPORT	AS REQUIRED	2.27	5.0
DAMPER	1	2.72	6.0
FORWARD OMNI SUPPORT	1	0.23	0.5
AFT OMNI SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	1.36	3.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	4.08	9.0
BALANCE WEIGHT PROVISION		5.4	12.0
• SPACECRAFT LESS SCIENCE		223.1	491.9

Despun Reflector, Option 1 (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
SCIENTIFIC INSTRUMENTS		45.41	100.1
MAGNETOMETER	1	4.04	8.9
ELECTRON TEMPERATURE PROBE	1	1.59	3.5
NEUTRAL MASS SPECTROMETER	1	6.26	13.8
ION MASS SPECTROMETER	1	1.68	3.7
UV SPECTROMETER	1	6.26	13.8
SOLAR WIND ANALYZER	1	5.76	12.7
IR RADIOMETER	1	6.26	13.8
X-BAND OCCULTATION	1	3.13	6.9
RF ALTIMETER	1	10.43	23.0
• SPACECRAFT (DRY)		268.5	592.0
PROPELLANTS AND PRESSURANT		162.6	358.4
INSERTION PROPELLANT AND EXPENDED INERTS		144.47	318.5
HYDRAZINE PROPELLANT		16.33	36.0
NITROGEN PRESSURANT		1.77	3.9
• SPACECRAFT LESS CONTINGENCY		431.1	950.4
CONTINGENCY (NET ALLOWABLE)		76.9	169.6
• GROSS SPACECRAFT AFTER SEPARATION		508.0	1120.0

APPENDIX 6E

MASS PROPERTIES
PRELIMINARY CONTINGENCY ANALYSIS

APPENDIX 6E

MASS PROPERTIES PRELIMINARY CONTINGENCY ANALYSIS

1. SUMMARY

Preliminary investigations were conducted to assess the magnitude of the overall weight contingency requirements for the probe bus, the orbiter, and the probes. The following briefly summarizes the overall contingency factors resulting from the analyses.

Item	Estimated Contingency Factor	Table Reference
Thor/Delta Configuration		
Probe bus	10.6% of dry bus weight	6E-2
Orbiter	10.2% of dry weight	6E-2
Probes	12.4% of total probe weight	6E-4
Atlas/Centaur Configurations		
Probe bus	13.7% of dry bus weight	6E-5
Orbiter	13.2% of dry weight	6E-5
Probes	16.5% of total probe weight	6E-5

2. ESTIMATING METHODOLOGY

For the Thor/Delta probe bus and orbiter configurations, the estimated contingency factors were based on an average of three estimating methods. The methods and the values used in the analyses are summarized in Table 6E-1. Results of the three approaches are summarized in Table 6E-2.

Since method 1 closely approximated the average value of the three methods, this method was used to estimate the Thor/Delta probe contingency. Values used in the analyses and the results are tabulated in Tables 6E-3 and 6E-4, respectively.

Table 6E-1. Weight Contingency Estimation Methodology, Probe Bus and Orbiter Spacecraft, Version III Science Payload

METHOD I. BASED ON APPLYING SUBSYSTEM LEVEL CONTINGENCY FACTORS AS DERIVED FROM A PRIORI TRW-DEVELOPED SPACECRAFT DATA.

SUBSYSTEM	CONTINGENCY FACTOR (%)
ELECTRICAL POWER	5
COMMUNICATIONS, ELECTRONICS	25
COMMUNICATIONS, ANTENNAS	15
ELECTRICAL DISTRIBUTION	16
DATA HANDLING	5
ATTITUDE CONTROL	15
PROPULSION (DRY)	2
THERMAL CONTROL	25
STRUCTURE	8
EXPERIMENTS	10

METHOD II. APPLYING UNIT LEVEL CONTINGENCY FACTORS BASED ON THE DEVELOPMENT STATUS OF THE UNITS AS SHOWN IN VOLUME III, SECTION 4.

STATUS OF UNITS	CODE (REFERENCE)	CONTINGENCY FACTOR (%)
EXISTING	1	2
MINIMUM MODIFIED EXISTING	2	5
MODERATELY MODIFIED EXISTING	3	8
NEW	4	15
EXPERIMENTS	3/4	11.5

METHOD III. BASED ON RSS METHOD OF ELEMENTS WITHIN EACH SUBSYSTEM LEVEL.

	FACTORS (%)
STRUCTURAL ELEMENTS	18
THERMAL CONTROL ELEMENTS	18
PROPULSION HARDWARE ELEMENTS	18
ELECTRONICS	
0 TO 4.6 KG (0 TO 10 LB) ITEMS	30
4.9 TO 13.6 KG (11 TO 30 LB) ITEMS	20
OVER 14.1 KG (31 LB) ITEMS	15
HARNESS/CONNECTORS	25
EXPERIMENTS	30

For the Atlas/Centaur probe bus and orbiter configurations, the same approach was used as in the Thor/Delta analysis. However, since the Atlas/Centaur configurations have not been investigated to the same level of detail as the Thor/Delta configurations, an additional uncertainty factor of 33 percent was applied to the resulting contingency values. Results of this analysis are presented in Table 6E-5.

As for the Atlas/Centaur probes, the same rationale was assumed. Thus the contingency factor estimated for the Thor/Delta probes was increased 33 percent, that is, from 12.4 percent to 16.5 percent.

Table 6E-2. Weight Contingency Estimate, Thor/Delta Probe Bus and Orbiter Spacecraft, Version III Science Payload

DESCRIPTION	WEIGHT [KG (LB)]	CONTINGENCY ESTIMATE [KG (LB)]			
		METHOD I	METHOD II	METHOD III	AVERAGE
PROBE BUS, DRY					
ELECTRICAL POWER	20.1 (44.3)	0.99 (2.2)	2.09 (4.6)	1.99 (4.4)	1.68 3.7)
COMMUNICATIONS	8.03 (17.7)	1.86 (4.1)	0.36 (0.8)	0.68 (1.5)	0.95 (2.1)
ELECTRICAL DISTRIBUTION	13.06 (28.8)	2.09 (4.6)	1.54 (3.4)	2.58 (5.7)	2.09 (4.6)
DATA HANDLING	3.85 (8.5)	0.18 (0.4)	0.09 (0.2)	0.15 (2.1)	0.41 (0.9)
ATTITUDE CONTROL	2.45 (5.4)	0.36 (0.8)	0.14 (0.3)	0.59 (1.3)	0.36 (0.8)
PROPULSION (DRY)	5.67 (12.5)	0.14 (0.3)	0.36 (0.8)	0.32 (0.7)	0.27 (0.6)
THERMAL CONTROL	9.75 (21.5)	2.45 (5.4)	0.09 (2.8)	1.09 (2.4)	1.59 (3.5)
STRUCTURE	44.08 (97.2)	3.54 (7.8)	6.39 (14.1)	2.13 (4.7)	4.04 (8.9)
EXPERIMENTS	11.16 (24.6)	1.13 (2.5)	1.27 (2.8)	1.77 (3.9)	1.41 (3.1)
BALANCE WEIGHTS	2.72 (6.0)	-	-	-	-
Σ	120.86 (266.5)	12.47 (28.1)	12.51 (29.8)	12.11 (26.7)	12.79 (28.2)
(%)		(10.5%)	(11.2%)	(10.0%)	(10.6%)
			(SELECTED PROBE BUS CONTINGENCY FACTOR)		
ORBITER SPACECRAFT, DRY					
ELECTRICAL POWER	38.14 (84.1)	1.90 (4.2)	4.53 (10.0)	3.17 (7.0)	3.22 (7.1)
COMMUNICATIONS	9.84 (21.7)	2.22 (4.9)	0.59 (1.3)	0.95 (2.1)	1.27 (2.8)
ELECTRICAL DISTRIBUTION	13.06 (28.8)	2.06 (4.6)	1.54 (3.4)	2.58 (5.7)	2.09 (4.6)
DATA HANDLING	5.67 (12.5)	0.27 (0.6)	0.36 (0.8)	0.99 (2.2)	1.54 (1.2)
ATTITUDE CONTROL	13.11 (28.9)	1.95 (4.3)	1.18 (2.6)	1.86 (4.1)	1.68 (3.7)
PROPULSION (DRY)	5.67 (12.5)	0.14 (0.3)	0.36 (0.8)	0.32 (0.7)	0.27 (0.6)
INSERTION MOTOR (BURNOUT)	9.07 (20.0)	0.18 (0.4)	0.18 (0.4)	0.36 (0.8)	0.23 (0.5)
THERMAL CONTROL	10.57 (23.3)	2.63 (5.8)	1.27 (2.8)	1.04 (2.3)	1.63 (3.6)
STRUCTURE	40.86 (90.1)	3.26 (7.2)	5.94 (13.1)	1.95 (4.3)	3.72 (8.2)
EXPERIMENTS	28.34 (62.5)	2.86 (6.3)	3.26 (7.2)	3.85 (8.5)	3.31 (7.3)
BALANCE WEIGHTS	2.72 (6.0)	-	-	-	-
Σ	177.05 (390.4)	17.5 (38.6)	19.22 (42.4)	17.09 (37.7)	17.96 (39.6)
(%)		(9.9%)	(10.9%)	(9.7%)	(10.2%)
			(SELECTED ORBITER SPACECRAFT CONTINGENCY FACTOR)		

Table 6E-3. Weight Contingency Estimation Factors, Large and Small Probes (Thor/Delta), Version III Science Payload

ITEM	CONTINGENCY FACTOR (%)	
	CASE A	CASE B
AEROSHELL*	8	8
HEATSHIELD	8	12
PRESSURE VESSEL	8	12
AUXILIARY STRUCTURE*	8	8
PARACHUTE	8	12
THERMAL CONTROL*	25	25
POWER	5	7.5
CABLING	16	24
DATA HANDLING	5	7.5
COMMUNICATIONS	15	22.5
ORDNANCE	16	24
SCIENCE*	10	10

NOTE: THE FOLLOWING FACTORS WERE APPLIED AT THE SUBSYSTEM LEVEL. CASE A REPRESENTS THE USE OF THE SAME CONTINGENCY FACTORS AS APPLIED IN THE PROBE BUS ANALYSIS. CASE B ASSUMES THE SAME FACTORS AS CASE A FOR THE ASTERISKED ITEMS; HOWEVER, FOR ALL OTHER ITEMS, THE CONTINGENCY FACTORS WERE INCREASED 50 PERCENT BECAUSE OF THE COMPLEXITY OF THE PROBE DESIGN.

Table 6E-4. Weight Contingency Estimate, Large and Small Probes (Thor/Delta), Version III Science Payload

ITEM	SMALL PROBE WEIGHT [KG (LB)]	SMALL PROBE CONTINGENCY [KG (LB)] CASE A	SMALL PROBE CONTINGENCY [KG (LB)] CASE B	LARGE PROBE WEIGHT [KG (LB)]	LARGE PROBE CONTINGENCY [KG (LB)] CASE A	LARGE PROBE CONTINGENCY [KG (LB)] CASE B
AEROSHELL	3.86 (8.51)	0.31 (0.68)	0.31 (0.68)	22.80 (50.29)	1.82 (4.02)	1.82 (4.02)
HEATSHIELD	4.03 (8.88)	0.32 (0.71)	0.49 (0.07)	19.18 (42.30)	1.53 (3.38)	2.30 (5.08)
PRESSURE VESSEL	4.31 (9.50)	0.34 (0.76)	0.52 (1.14)	23.49 (51.80)	1.88 (4.14)	2.82 (6.22)
AUXILIARY STRUCTURE	-	-	-	10.37 (22.88)	0.83 (1.83)	0.83 (1.83)
PARACHUTE	-	-	-	5.89 (13.00)	0.47 (1.04)	0.70 (1.56)
THERMAL CONTROL	3.83 (8.45)	0.96 (7.11)	0.96 (2.11)	6.66 (14.70)	1.67 (3.68)	1.67 (3.68)
POWER	4.08 (9.00)	0.20 (0.45)	0.31 (0.68)	7.84 (17.30)	0.39 (0.87)	0.59 (1.30)
CABLING	0.95 (2.10)	0.15 (0.34)	0.23 (0.50)	6.80 (15.00)	1.09 (7.40)	1.63 (3.60)
DATA HANDLING	1.45 (3.20)	0.07 (0.16)	0.11 (0.24)	2.27 (5.00)	0.11 (0.25)	0.17 (0.38)
COMMUNICATIONS	1.09 (2.40)	0.16 (0.36)	0.24 (0.54)	5.17 (11.40)	0.76 (1.71)	1.16 (2.57)
ORDNANCE	0.1 (0.21)	0.01 (0.03)	0.02 (0.05)	2.99 (6.60)	0.48 (1.06)	0.72 (1.58)
SCIENCE	2.22 (4.90)	0.22 (0.49)	0.22 (0.49)	24.03 (53.00)	2.40 (5.03)	2.40 (5.30)
BALANCE WEIGHT	0.23 (0.50)	-	-	1.59 (3.5)	-	-
Σ	26.14 (57.65) (EACH)	7.76 (6.09)	3.40 (7.50)	139.08 (306.77)	13.46 (29.68)	16.83 (37.12)
(%)	78.42 (172.95) (TOTAL)	8.28 (18.27)	10.20 (22.50)			
		(10.6%)	(13%)		(9.7%)	(12.1%)

21.74 KG (47.95 LB)
(10.0%)

27.03 (59.62 LB)
(12.4%)

(SELECTED OVERALL PROBE CONTINGENCY FACTOR)

Table 6E-5. Weight Contingency Estimate-Atlas/Centaur Probe Bus, Orbiter, and Probes, Version III Science Payload

DESCRIPTION	WEIGHT [KG (LB)]	CONTINGENCY ESTIMATE [KG (LB)] METHOD I	METHOD II	METHOD III	AVERAGE
PROBE BUS, DRY					
ELECTRICAL POWER	20.1 (44.3)	0.99 (2.2)	2.13 (4.7)	1.99 (4.4)	1.72 (3.8)
COMMUNICATIONS	12.43 (27.4	2.99 (6.6)	0.41 (0.9)	1.27 (2.8)	1.54 (3.4)
ELECTRICAL DISTRIBUTION	15.33 (33.8)	2.45 (5.4)	1.91 (4.2)	3.08 (6.8)	2.49 (5.5)
DATA HANDLING	3.85 (8.5)	0.18 (0.4)	0.09 (0.2)	0.95 (2.1)	0.41 (0.9)
ATTITUDE CONTROL	2.45 (5.4)	0.36 (0.8)	0.14 (0.3)	0.59 (1.3)	0.36 (0.8)
PROPULSION (DRY)	5.12 (13.5)	0.14 (0.3)	0.41 (0.9)	0.36 (0.8)	0.32 (0.7)
THERMAL CONTROL	14.97 (33.0)	3.76 (8.3)	2.09 (4.6)	1.91 (4.2)	2.59 (5.7)
STRUCTURE	88.99 (169.2)	6.12 (13.5)	10.02 (22.1)	3.63 (8.0)	6.58 (14.5)
EXPERIMENTS	11.16 (24.6)	1.18 (2.6)	1.36 (3.0)	1.95 (4.3)	1.50 (3.3)
BALANCE WEIGHT	5.44 (12.0)	-	-	-	-
Σ	169.42 (373.5)	18.19 (40.1)	18.55 (40.9)	15.74 (34.7)	15.51 (38.6)
(%)		(10.7%)	(11.0%)	(9.3%)	(10.3%)*

*SINCE THE ATLAS/CENTAUR CONFIGURATIONS HAVE NOT BEEN INVESTIGATED TO THE SAME LEVEL OF DETAIL AS THOSE FOR THOR/DELTA, AN ADDITION UNCERTAINTY FACTOR OF 33 PERCENT IS ASSUMED. OVERALL CONTINGENCY FACTOR ASSUMED 13.7 PERCENT.

DESCRIPTION	WEIGHT [KG (LB)]	METHOD I	METHOD II	METHOD III	AVERAGE
ORBITER SPACECRAFT, DRY					
ELECTRICAL POWER	38.14 (84.1)	1.90 (4.2)	4.53 (10.1)	3.13 (6.9)	3.18 (7.0)
COMMUNICATIONS	14.24 (31.4)	3.31 (7.3)	0.68 (1.5)	1.41 (3.1)	1.81 (4.0)
ELECTRICAL DISTRIBUTION	15.33 (33.8)	2.45 (5.4)	1.91 (4.2)	3.08 (6.8)	2.49 (5.5)
DATA HANDLING	5.67 (12.5)	0.27 (0.6)	0.36 (0.8)	0.99 (2.2)	0.54 (1.2)
ATTITUDE CONTROL	13.11 (28.9)	1.95 (4.3)	1.18 (2.6)	1.86 (4.1)	1.78 (3.7)
PROPULSION (DRY)	6.12 (13.5)	0.14 (0.3)	0.41 (0.9)	0.36 (0.8)	0.32 (0.7)
INSERTION MOTOR (BURNOUT)	18.64 (41.1)	0.36 (0.8)	0.36 (0.8)	0.73 (1.6)	4.53 (1.0)
THERMAL CONTROL	15.01 (33.1)	3.76 (8.3)	1.96 (4.3)	1.59 (3.5)	2.45 (5.4)
STRUCTURE	71.07 (156.7)	5.67 (12.5)	10.43 (23.0)	3.31 (7.3)	6.49 (14.3)
EXPERIMENTS	33.06 (12.0)	-	-	-	-
Σ	235.87 (520.0)	23.13 (51.0)	25.67 (56.6)	21.23 (46.8)	23.36 (51.5)
(%)		(9.8%)	(10.9%)	(9.0%)	(9.9%)*

*WITH ADDITIONAL UNCERTAINTY FACTOR OF 33 PERCENT, OVERALL CONTINGENCY FACTOR ASSUMED 13.2 PERCENT

PROBES

ASSUMED 33 PERCENT GREATER THAN THOR/DELTA. OVERALL CONTINGENCY FACTOR ASSUMED 16.5 PERCENT.

APPENDIX 6F

DETAILED MASS PROPERTIES
OPTIONAL ATLAS/CENTAUR ORBITER CONFIGURATIONS, VERSION III SCIENCE PAYLOAD

APPENDIX 6F

DETAILED MASS PROPERTIES OPTIONAL ATLAS/CENTAUR ORBITER CONFIGURATIONS, VERSION III SCIENCE PAYLOAD

Atlas/Centaur Orbiter, 12-Watt Fanbeam, Fanscan Configuration, Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
ELECTRICAL POWER		35.7	78.6
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	11.43	25.2
BATTERY (16-12 AH NI-CD CELLS)	1	10.48	23.1
POWER CONTROL UNIT	1	6.35	14.0
CTRF/INVERTER	1	7.39	16.3
COMMUNICATIONS		11.8	25.9
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
POWER AMPLIFIER	4	1.09	2.4
TRANSMITTER DRIVERS	2	1.09	2.4
HYBRIDS	5	0.23	0.5
DIPLEXERS	2	1.95	4.3
SWITCHES	5	1.36	3.0
FORWARD OMNI ANTENNA	1	0.14	0.3
AFT OMNI ANTENNA	1	0.23	0.5
FANBEAM ANTENNA	1	1.13	2.5
FANSCAN ANTENNA	1	0.45	1.0
RF CABLE AND CONNECTORS	AS REQUIRED	1.36	3.0
ELECTRICAL DISTRIBUTION		15.8	34.8
COMMAND DISTRIBUTION UNIT	1	4.45	9.8
HARNESS AND CONNECTORS	AS REQUIRED	11.34	25.0
DATA HANDLING		12.5	27.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	8.62	19.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		2.7	6.0
CONTROL ELECTRONICS ASSEMBLY	1	2.31	5.1
SUN SENSOR ASSEMBLY	2	0.41	0.9
PROPULSION		6.9	15.3
THRUSTERS	8	2.18	4.8
PROPELLANT TANK	3	3.13	6.9
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATER AND MISCELLANEOUS	AS REQUIRED	1.09	2.4

Atlas/Centaur Orbiter, 12-Watt Fanbeam, Fanscan Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
SOLID INSERTION MOTOR (BURNOUT)	1	18.7	41.0
THERMAL CONTROL		15.0	33.0
INSULATION	AS REQUIRED	8.12	17.9
FORWARD CLOSURE	1	1.32	2.9
AFT CLOSURE	1	0.54	1.2
LOUVERS	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	1.26	3.0
STRUCTURE		73.0	161.1
CENTRAL CYLINDER ASSEMBLY		22.77	50.2
UPPER RING	1		(6.4)
CYLINDER	1		(6.6)
PLATFORM RINGS	2		(3.4)
LOWER FRUSTUM	1		(11.2)
SEPARATION RING	1		(14.6)
SRM MOUNTING RING	1		(6.5)
ATTACH HARDWARE	AS REQUIRED		(1.5)
PLATFORM/COMPARTMENT ASSEMBLY		27.03	59.6
UPPER STRUTS	15		(5.7)
PLATFORM STRUTS	9		(3.7)
VERTICALS	9		(4.0)
UPPER RING ASSEMBLY	1		(5.6)
PLATFORM STRUT FITTINGS	9		(1.8)
PLATFORM ASSEMBLY	1		(34.6)
BRACKET AND ATTACH HARDWARE	AS REQUIRED		(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY		5.81	12.8
UPPER RING	1		(3.4)
LOWER RING	1		(4.0)
STRUTS	18		(5.4)
ANTENNA SUPPORT ASSEMBLY	1	2.59	5.7
MAGNETOMETER BOOM ASSEMBLY	1	4.22	9.3
PROPULSION SUPPORT	AS REQUIRED	2.27	5.0
DAMPER	1	2.72	6.0
AFT OMNI SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	1.36	3.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	4.08	9.0

Atlas/Centaur Orbiter, 12-Watt Fanbeam, Fanscan Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
BALANCE WEIGHT PROVISION		5.4	12.0
• SPACECRAFT LESS SCIENCE		197.5	435.3
SCIENTIFIC INSTRUMENTS		33.0	72.9
MAGNETOMETER	1	2.49	5.5
ELECTRON TEMPERATURE PROBE	1	1.00	2.2
NEUTRAL MASS SPECTROMETER	1	5.44	12.0
ION MASS SPECTROMETER	1	1.45	3.2
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.54	10.0
RF ALTIMETER	1	12.70	28.0
• SPACECRAFT (DRY)		230.5	508.2
PROPELLANTS AND PRESSURANT		140.3	309.4
INSERTION PROPELLANT AND EXPENDED INERTS		126.10	278.0
HYDRAZINE PROPELLANT		13.97	30.8
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		370.8	817.6
CONTINGENCY (NET ALLOWABLE)		64.6	142.4
• GROSS SPACECRAFT AFTER SEPARATION		435.4	960.0

Atlas/Centaur Orbiter, Earth-Pointing Configuration,
Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
ELECTRICAL POWER		35.7	78.6
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	11.43	25.2
BATTERY (16-12 AH NI-CD CELLS)	1	10.48	23.1
POWER CONTROL UNIT	1	6.35	14.0
CTRF/INVERTER	1	7.39	16.3
COMMUNICATIONS		15.7	34.5
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
POWER AMPLIFIER	2	0.54	1.2
TRANSMITTER DRIVERS	2	1.09	2.4
HYBRIDS	2	0.09	0.2
DIPLEXERS	2	1.95	4.3
SWITCHES	7	1.91	4.2
FORWARD OMNI ANTENNA	1	0.41	0.9
AFT OMNI ANTENNA	1	0.14	0.3
MEDIUM-GAIN ANTENNA	1	0.91	2.0
HIGH-GAIN ANTENNA ASSEMBLY	1	4.54	10.0
RF CABLE AND CONNECTORS	AS REQUIRED	1.36	3.0
ELECTRICAL DISTRIBUTION		15.8	34.8
COMMAND DISTRIBUTION UNIT	1	4.45	9.8
HARNESS AND CONNECTORS	AS REQUIRED	11.34	25.0
DATA HANDLING		12.5	27.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	8.62	19.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		2.7	6.0
CONTROL ELECTRONICS ASSEMBLY	1	2.31	5.1
SUN SENSOR ASSEMBLY	2	0.41	0.9
PROPULSION		6.9	15.3
THRUSTERS	8	2.18	4.8
PROPELLANT TANK	3	3.13	6.9
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATER AND MISCELLANEOUS	AS REQUIRED	1.09	2.4

Atlas/Centaur Orbiter, Earth-Pointing Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
SOLID INSERTION MOTOR (BURNOUT)	1	18.7	41.1
THERMAL CONTROL		15.0	33.0
INSULATION	AS REQUIRED	8.12	17.9
FORWARD CLOSURE	1	1.32	2.9
AFT CLOSURE	1	0.54	1.2
LOUVERS	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	1.36	3.0
STRUCTURE		75.4	166.4
CENTRAL CYLINDER ASSEMBLY		22.77	50.2
UPPER RING	1		(6.4)
CYLINDER	1		(6.6)
PLATFORM RINGS	2		(3.4)
LOWER FRUSTUM	1		(11.2)
SEPARATION RING	1		(14.6)
SRM MOUNTING RING	1		(6.5)
ATTACH HARDWARE	AS REQUIRED		(1.5)
PLATFORM/COMPARTMENT ASSEMBLY		27.03	59.6
UPPER STRUTS	15		(5.7)
PLATFORM STRUTS	9		(3.7)
VERTICALS	9		(4.0)
UPPER RING ASSEMBLY	1		(5.6)
PLATFORM STRUT FITTINGS	9		(1.8)
PLATFORM ASSEMBLY	1		(34.6)
BRACKET AND ATTACH HARDWARE	AS REQUIRED		(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY		5.81	12.8
UPPER RING	1		(3.4)
LOWER RING	1		(4.0)
STRUTS	18		(5.4)
NM/IM SPECTROMETER BOOM ASSEMBLY	1	4.54	10.0
MAGNETOMETER BOOM ASSEMBLY	1	4.22	9.3
PROPULSION SUPPORT	AS REQUIRED	2.27	5.0
DAMPER	1	2.72	6.0
FORWARD OMNI SUPPORT	1	0.23	0.5
AFT OMNI SUPPORT	1	0.23	0.5
MEDIUM-GAIN ANTENNA SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	1.36	3.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	4.08	9.0

Atlas/Centaur Orbiter, Earth-Pointing Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
BALANCE WEIGHT PROVISION		5.4	12.0
• SPACECRAFT BUS LESS SCIENCE		203.8	449.2
SCIENTIFIC INSTRUMENTS		33.0	72.9
MAGNETOMETER	1	2.49	5.5
ELECTRON TEMPERATURE PROBE	1	1.00	2.2
NEUTRAL MASS SPECTROMETER	1	5.44	12.0
ION MASS SPECTROMETER	1	1.45	3.2
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.54	10.0
RF ALTIMETER	1	12.70	28.0
• SPACECRAFT (DRY)		236.8	522.1
PROPELLANTS AND PRESSURANT		140.3	309.4
INSERTION PROPELLANT AND EXPENDED INERTS		126.10	278.0
HYDRAZINE PROPELLANT		13.97	30.8
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		377.1	831.5
CONTINGENCY (NET ALLOWABLE)		58.3	128.5
• GROSS SPACECRAFT AFTER SEPARATION		435.4	960.0

Atlas/Centaur Orbiter, Despun Reflector Configuration, Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
ELECTRICAL POWER		40.8	89.9
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	12.34	27.2
BATTERY (16-15 AH NI-CD CELLS)	1	13.06	28.8
POWER CONTROL UNIT	1	6.35	14.0
CTRF/INVERTER	1	9.03	19.9
COMMUNICATIONS		12.7	27.9
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
POWER AMPLIFIER	4	1.09	2.4
TRANSMITTER DRIVERS	2	1.09	2.4
HYBRIDS	5	0.23	0.5
DIPLEXERS	2	1.95	4.3
SWITCHES	5	1.36	3.0
FORWARD OMNI ANTENNA	1	0.14	0.3
AFT OMNI ANTENNA	1	0.23	0.5
FANSCAN ANTENNA	1	0.45	1.0
HIGH-GAIN ANTENNA ASSEMBLY	1	2.04	4.5
RF CABLE AND CONNECTORS	AS REQUIRED	1.36	3.0
ELECTRICAL DISTRIBUTION		15.8	34.8
COMMAND DISTRIBUTION UNIT	1	4.45	9.8
HARNESS AND CONNECTORS	AS REQUIRED	11.34	25.0
DATA HANDLING		12.5	27.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	8.62	19.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		12.8	28.3
CONTROL ELECTRONICS ASSEMBLY	1	2.31	5.1
SUN SENSOR ASSEMBLY	2	0.41	0.9
DESPIN CONTROL ASSEMBLY	2	3.63	8.0
DESPIN DRIVE ASSEMBLY	1	6.49	14.3
PROPULSION		6.9	15.3
THRUSTERS	8	2.18	4.8
PROPELLANT TANK	3	3.13	6.9
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATER AND MISCELLANEOUS	AS REQUIRED	1.09	2.4

Atlas/Centaur Orbiter, Despun Reflector Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
SOLID INSERTION MOTOR (BURNOUT)	1	18.7	41.1
THERMAL CONTROL		15.0	33.0
INSULATION	AS REQUIRED	8.12	17.9
FORWARD CLOSURE	1	1.32	2.9
AFT CLOSURE	1	0.54	1.2
LOUVERS	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	1.36	3.0
STRUCTURE		72.0	158.8
CENTRAL CYLINDER ASSEMBLY		22.77	50.2
UPPER RING	1		(6.4)
CYLINDER	1		(6.6)
PLATFORM RINGS	2		(3.4)
LOWER FRUSTUM	1		(11.2)
SEPARATION RING	1		(14.6)
SRM MOUNTING RING	1		(6.5)
ATTACH HARDWARE	AS REQUIRED		(1.5)
PLATFORM/COMPARTMENT ASSEMBLY		27.03	59.6
UPPER STRUTS	15		(5.7)
PLATFORM STRUTS	9		(3.7)
VERTICALS	9		(4.0)
UPPER RING ASSEMBLY	1		(5.6)
PLATFORM STRUT FITTINGS	9		(1.8)
PLATFORM ASSEMBLY	1		(34.6)
BRACKET AND ATTACH HARDWARE	AS REQUIRED		(4.2)
SOLAR ARRAY SUPPORT ASSEMBLY		5.81	12.8
UPPER RING	1		(3.4)
LOWER RING	1		(4.0)
STRUTS	18		(5.4)
ANTENNA SUPPORT ASSEMBLY	1	1.32	2.9
MAGNETOMETER BOOM ASSEMBLY	1	4.22	9.3
PROPULSION SUPPORT	AS REQUIRED	2.27	5.0
DAMPER	1	2.72	6.0
FORWARD OMNI SUPPORT	1	0.23	0.5
AFT OMNI SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	1.36	3.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	4.08	9.0

Atlas/Centaur Orbiter, Despun Reflector Configuration, Version III Science Payload (Continued)

	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
BALANCE WEIGHT PROVISION		5.4	12.0
• SPACECRAFT LESS SCIENCE		212.6	468.6
SCIENTIFIC INSTRUMENTS		33.0	72.9
MAGNETOMETER	1	2.49	5.5
ELECTRON TEMPERATURE PROBE	1	1.00	2.2
NEUTRAL MASS SPECTROMETER	1	5.44	12.0
ION MASS SPECTROMETER	1	1.45	3.2
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.54	10.0
RF ALTIMETER	1	12.70	28.0
• SPACECRAFT (DRY)		245.6	541.5
PROPELLANTS AND PRESSURANT		140.3	309.4
INSERTION PROPELLANT AND EXPENDED INERTS		126.10	278.0
HYDRAZINE PROPELLANT		13.97	30.8
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		385.9	850.9
CONTINGENCY (NET ALLOWABLE)		49.5	109.1
• GROSS SPACECRAFT AFTER SEPARATION		435.4	960.0

APPENDIX 6G

MASS PROPERTIES
PRELIMINARY UNCERTAINTY ANALYSES
VERSION III SCIENCE PAYLOAD

APPENDIX 6G

MASS PROPERTIES PRELIMINARY UNCERTAINTY ANALYSES VERSION III SCIENCE PAYLOAD

1. ATLAS/CENTAUR PROBE MISSION

A preliminary uncertainty analysis to determine allowable mass properties deviations in critical components is presented for the Atlas/Centaur probe mission.

The limiting condition is assumed to be the attitude control system requirement of [≤0.0035 rad (≤0.2 deg)] principal axis misalignment with the geometric longitudinal axis. The allowable products of inertia for the current spacecraft configuration are:

$$P_{xy} = (3886\ \text{kg}\cdot\text{cm}^2)\ (0.0035\ \text{rad})$$

$$P_{xy} = (1328\ \text{lb-in.}^2)\ (0.0035\ \text{rad})$$

$$P_{xz} = (4978\ \text{kg}\cdot\text{cm}^2)\ (0.0035\ \text{rad})$$

$$P_{xz} = (1701\ \text{lb-in.}^2)\ (0.0035\ \text{rad})$$

The recommended allocations for probe mass properties are as follows:

	Large Probe	Small Probe
Weight uncertainty and variation from nominal	≤ 0.91 kg (≤ 2.0 lb)	≤ 0.23 kg (≤ 0.5 lb)
Radial center-of-gravity offset from probe centerline	≤ 0.127 cm (≤ 0.050 in.)	≤ 0.127 cm (≤ 0.050 in.)
Longitudinal center-of-gravity uncertainty from nominal	≤ 0.762 cm (≤ 0.30 in.)	≤ 0.127 cm (≤ 0.050 in.)
Product of inertia in plane of longitudinal axis	≤293 kg · cm^2 (≤100 lb-in.^{2})	≤103 kg · cm^2) (≤35 lb-in.^{2})

The steps necessary to verify the spacecraft mass properties are:

- Dynamically balance the probe bus in the dry condition
- Dynamically balance each probe
- Integrate probes with bus and add propellant and pressurant with no further balancing required.

The effects of tolerance allocations on the spacecraft mass properties are summarized in Table 6G-1. Detailed effects of variations in large and small probe mass properties are presented in Table 6G-2.

Table 6G-1. Atlas/Centaur Probe Spacecraft Dynamic Balance Tolerance Allocations (Stowed Condition)

SOURCE	TOLERANCE (LB-IN.^{2}) P_{XY}	P_{XZ}	COMMENTS
BALANCE (MACHINE ACCURACY)	100	100	AVERAGE BASED ON CLOCKWISE AND COUNTERCLOCKWISE BALANCING
PROPELLANT WEIGHT AND C.G. (TANK 1)	54	177	±0.50 INCH UNCERTAINTY IN RADIAL AND LONGITUDINAL C.G. AND ±0.1 LB PROPELLANT WEIGHT UNCERTAINTY IN EACH TANK
PROPELLANT WEIGHT AND C.G. (TANK 2)	101	100	
PROPELLANT WEIGHT AND C.G. (TANK 3)	101	100	
POST ALIGNMENT EQUIPMENT CHANGES	40	40	
LARGE PROBE PRINCIPAL AXIS	100	100	±100 LB-IN.^{2} UNCERTAINTY IN BALANCE OF PROBE
LARGE PROBE C.G. ACCURACY	428	428	±0.050 INCH RADIAL C.G. UNCERTAINTY
LARGE PROBE ALIGNMENT ACCURACY	428	428	±0.05 INCH UNCERTAINTY IN ALIGNMENT RADIALLY AND LONGITUDINALLY
SMALL PROBES PRINCIPAL AXES	60	60	±35 LB-IN.^{2} EACH PROBE
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY	71	343	±0.50 LB WEIGHT UNCERTAINTY AND ±0.05 INCH LONGITUDINAL C.G. AND ±0.05 INCH RADIAL C.G.
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY	299	182	±0.50 LB WEIGHT UNCERTAINTY AND ±0.05 INCH LONGITUDINAL C.G. AND ±0.05 INCH RADIAL C.G.
SMALL PROBE NO. 3 WEIGHT AND C.G. ACCURACY	299	182	±0.50 LB WEIGHT UNCERTAINTY AND ±0.05 INCH LONGITUDINAL C.G. AND ±0.05 INCH RADIAL C.G.
SMALL PROBE NO. 1 ALIGNMENT ACCURACY	71	314	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 2 ALIGNMENT ACCURACY	299	169	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 3 ALIGNMENT ACCURACY	299	169	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
TOTAL WORST CASE	2750	2892	
TOTAL ROOT-SUM-SQUARE (RSS)	885	885	
LIMIT VALUE	1328	1701	±0.2 DEGREE UNCERTAINTY IN PRINCIPAL AXIS ALIGNMENT FOR ATTITUDE DETERMINATION AND PROBE DEPLOYMENT ERROR BUDGET
AVAILABLE MARGIN	443	816	USING RSS VALUES

Table 6G-2. Detailed Effects of Probe Tolerance Allocations

SOURCE	TOLERANCE (LB-IN.^{2}) P_{XY}	P_{XZ}	COMMENTS
LARGE PROBE PRINCIPAL AXIS	100	100	±100 LB-IN^2 UNCERTAINTY IN BALANCE OF PROBE
LARGE PROBE RADIAL C.G.	428	428	±0.050 INCH RADIAL C.G. UNCERTAINTY
LARGE PROBE LONGITUDINAL C.G.	--	--	±0.30 INCH LONGITUDINAL C.G. UNCERTAINTY
SMALL PROBE PRINCIPAL AXIS	60	60	±35 LB-IN.^{2} EACH PROBE (ROOT-SUM-SQUARE (RSS) TOTAL OF 60 LB-IN.^{2})
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY (RSS)	(71)	(343)	RSS OF A), B), AND C) BELOW
A) WEIGHT ACCURACY	0	138	±0.50 POUND UNCERTAINTY
B) RADIAL C.G. ACCURACY	71	71	±0.050 INCH RADIAL OFFSET FROM PROBE CENTERLINE
C) LONGITUDINAL C.G. ACCURACY	0	306	±0.050 INCH LONGITUDINAL C.G. VARIATION FROM NOMINAL
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY (RSS)	(299)	(182)	RSS OF A), B), AND C) BELOW
A) WEIGHT ACCURACY	120	69	±0.50 POUND UNCERTAINTY
B) RADIAL C.G. ACCURACY	71	71	±0.050 INCH RADIAL OFFSET FROM PROBE CENTERLINE
C) LONGITUDINAL C.G. ACCURACY	265	153	±0.050 INCH LONGITUDINAL C.G. VARIATION FROM NOMINAL
SMALL PROBE NO. 3 WEIGHT AND C.G.	(299)	(182)	RSS OF A), B), AND C) BELOW
A) WEIGHT ACCURACY	120	69	±0.050 POUND UNCERTAINTY
B) RADIAL C.G. ACCURACY	71	71	±0.050 INCH RADIAL OFFSET FROM PROBE CENTERLINE
C) LONGITUDINAL C.G. ACCURACY	265	153	±0.050 INCH LONGITUDINAL C.G. VARIATION FROM NOMINAL

C-2

2. THOR/DELTA PROBE MISSION

2.1 Introduction

In considering the Thor/Delta launch vehicle for the probe mission, certain mass properties limits must be imposed on the bus and probe mass properties to meet the overall requirements of the launch vehicle and the mission performance requirements. This study is directed primarily at the large and small probe requirements as they impact the overall payload mass properties.

Two sets of requirements were addressed. The requirements as specified by the customer (Reference 1) are significantly more stringent than those stipulated by the launch vehicle manufacturer (Reference 2). The following results summarize the probe mass properties requirements for the Thor/Delta launch consideration.

A. Spacecraft Requirements (Reference 1)

Center-of-gravity offset from spacecraft centerline		≤0.038 cm (≤0.015 in.)
Principal axis (roll) misalignment with spacecraft geometric axis		≤0.002 rad (marginal)
Probe limits:	Large Probe	Small Probe
Weight uncertainty and variation from nominal	≤0.91 kg (≤2.0 lb)	≤0.05 kg (≤0.10 lb)
Radial center-of-gravity offset from probe centerline	≤0.025 cm (≤0.010 in.)	≤0.025 cm (≤0.010 in.)
Longitudinal center-of-gravity variation from nominal value	≤0.762 cm (≤0.30 in.)	≤0.254 cm (≤0.10 in.)
Product of inertia in plane of longitudinal axis	≤38 kg · cm^2 (≤13 lb-in.^{2})	≤2.9 kg · cm^2 (≤1.0 lb-in.^{2})

B. Spacecraft Requirements (Reference 2)

Center-of-gravity offset from spacecraft centerline		≤0.127 cm (≤0.050 in.)
Principal axis (roll) misalignment with spacecraft geometric axis		≤0.020 rad
Probe limits:	Large Probe	Small Probe
Weight uncertainty and variation from nominal	≤0.91 kg (≤2.0 lb)	≤0.23 kg (≤0.5 lb)
Radial center-of-gravity offset from probe centerline	≤0.127 cm (≤0.050 in.)	≤0.076 cm (≤0.030 in.)
Longitudinal center-of-gravity uncertainty from nominal value	≤0.762 cm (≤0.30 in.)	≤0.508 cm (≤0.20 in.)
Product of inertia in plane of longitudinal axis	≤293 kg • cm^2 (≤100 lb-$in.^2$)	≤29 kg • cm^2 (≤10 lb-$in.^2$)

2.3 Discussion

The steps necessary to verify the spacecraft mass properties are:

- Dynamically balance the probe bus in the dry condition
- Dynamically balance each probe
- Integrate probes with bus and add propellant and pressurant with no further balacing required.

The significant mass properties assumed for this study are as follows:

	WEIGHT [KG (LB)]	LONGITUDINAL* CENTER OF GRAVITY [CM (IN.)]	RADIAL** CENTER OF GRAVITY [CM (IN.)]	MOMENTS OF INERTIA [KG·M2 (SLUG-FT2)]		
				I_{XX}(ROLL)	I_{YY}	I_{ZZ}
LARGE PROBE	143.2 (315.7)	95.3 (37.5)	0.0	16.29 (12.02)	13.19 (9.73)	11.9 (8.85)
SMALL PROBE	26.4 (58.1) EACH	25.4 (10.0)	81.3 (32.0)	0.412 (0.304)	0.358 (0.264)	0.358 (0.264)
SPACECRAFT AND LAUNCH	385.1 (849.0)	50.5 (19.9)	0.0	145.6 (107.4)	122.7 (90.5)	118.1 (87.1)
PROPELLANT AND PRESSURANT	6.49 (14.3) EACH	25.4 (10.0)	129.0 (50.8)	-	-	-

*REFERENCE IS SPACECRAFT SEPARATION PLANE

**RADIAL CENTER-OF-GRAVITY DISTANCE IS RADIAL DISTANCE FROM EACH ITEM CENTER-OF-GRAVITY TO THE SPACECRAFT CENTERLINE

The detailed lists of uncertainties assigned to each variable are included in Tables 6G-3 and 6G-4 for the looser requirements of 0.127 cm (0.050 in.) and 0.020 radian. The assignment of tighter tolerances as indicated in Tables 6G-5 and 6G-6 was made in an attempt to satisfy the requirements of 0.127 cm and 0.002 radian.

Another approach to satisfy the tighter requirements would be to balance the total spacecraft including probes, thus eliminating the uncertainties in probe mass properties and alignments in the launch condition. The 0.002 radian requirement could be met with this technique provided the balance machine accuracy of ±234 kg $\cdot$ cm^2 (±80 lb-in.^{2}) is realistic. However, the practicality of this approach is questionable regarding the probes remaining in place after the balancing operation and during launch preparation.

Table 6G-3. Thor/Delta Probe Spacecraft Static Balance Tolerance Allocations (Stowed Condition)

SOURCE	TOLERANCE (IN.-LB)	COMMENTS
BALANCE (MACHINE) ACCURACY	1.0	AVERAGE BASED ON CLOCKWISE AND COUNTERCLOCKWISE BALANCING
PROPELLANT WEIGHT AND C.G. (TANK 1) 14.3 POUNDS	2.1	ALIGNMENT OF TANKS (±0.05 INCH) AND PROPELLANT EQUAL LOADING ACCURACY (±0.1 POUND EACH TANK AT 20 INCH RADIUS)
PROPELLANT WEIGHT AND C.G. (TANK 2) 14.3 POUNDS	2.1	
PROPELLANT WEIGHT AND C.G. (TANK 3) 14.3 POUNDS	2.1	
SPACECRAFT CENTERLINE ALIGNMENT	8.0	±0.010 INCH ALIGNMENT REPEATABILITY (DRY SPACECRAFT) WITH BALANCER
POST-ALIGNMENT EQUIPMENT CHANGES	1.0	ALLOWANCE FOR EQUIPMENT CHANGES WITHOUT REBALANCING
SUBTOTAL WORST CASE	(16.3)	
SUBTOTAL ROOT-SUM-SQUARE (RSS)	(8.9)	
LARGE PROBE C.G. ACCURACY	15.8	±0.050 INCH RADIAL C.G. UNCERTAINTY
LARGE PROBE ALIGNMENT ACCURACY	15.8	±0.05 INCH ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY	16.1	±0.030 INCH RADIAL C.G. UNCERTAINTY AND ±0.50 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY	16.1	±0.030 INCH RADIAL C.G. UNCERTAINTY AND ±0.50 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 3 WEIGHT AND C.G. ACCURACY	16.1	±0.030 INCH RADIAL C.G. UNCERTAINTY AND ±0.50 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 1 ALIGNMENT ACCURACY	2.9	±0.05 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 2 ALIGNMENT ACCURACY	2.9	±0.05 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 3 ALIGNMENT ACCURACY	2.9	±0.05 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SUBTOTAL WORST CASE	(88.6)	
SUBTOTAL RSS	(36.1)	
TOTAL WORST CASE	104.9	
TOTAL RSS	37.2	
LIMIT VALUE (THOR/DELTA USER'S GUIDE)	42.5	±0.050 INCH UNCERTAINTY IN 849-POUND SPACECRAFT
AVAILABLE MARGIN	5.3	USING RSS VALUES

Table 6G-4. Thor/Delta Probe Spacecraft Dynamic Balance Tolerance Allocations (Stowed Condition)

SOURCE	TOLERANCE (LB-IN.^{2}) P_{XY}	P_{XZ}	COMMENTS
BALANCE (MACHINE ACCURACY)	80	80	AVERAGE BASED ON CLOCKWISE AND COUNTERCLOCKWISE BALANCING
PROPELLANT WEIGHT AND C.G. (TANK 1)	14	22	±0.05 INCH UNCERTAINTY IN RADIAL AND LONGITUDINAL C.G. AND ±0.1 POUND PROPELLANT WEIGHT UNCERTAINTY IN EACH TANK
PROPELLANT WEIGHT AND C.G. (TANK 2)	14	22	
PROPELLANT WEIGHT AND C.G. (TANK 3)	25	7	
POST-ALIGNMENT EQUIPMENT CHANGES	20	20	
SUBTOTAL WORST CASE	(153)	(151)	
SUBTOTAL ROOT-SUM-SQUARE (RSS)	(88)	(88)	
LARGE PROBE PRINCIPAL AXIS	100	100	±100 LB-IN.^{2} UNCERTAINTY IN BALANCE OF PROBE
LARGE PROBE C.G. ACCURACY	278	278	±0.050 INCH RADIAL C.G. UNCERTAINTY
LARGE PROBE ALIGNMENT ACCURACY	278	278	±0.05 INCH UNCERTAINTY IN ALIGNMENT RADIALLY AND LONGITUDINALLY
SMALL PROBES PRINCIPAL AXES	17	17	±10 LB-IN.^{2} EACH PROBE
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY	57	99	±0.50 POUND WEIGHT UNCERTAINTY AND ±0.20 INCH LONGITUDINAL C.G. AND ±0.030 INCH RADIAL C.G.
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY	57	99	±0.50 POUND WEIGHT UNCERTAINTY AND ±0.20 INCH LONGITUDINAL C.G. AND ±0.030 INCH RADIAL C.G.
SMALL PROBE NO. 3 WEIGHT AND C.G. ACCURACY	57	2	±0.50 POUND WEIGHT UNCERTAINTY AND ±0.20 INCH LONGITUDINAL C.G. AND ±0.030 INCH RADIAL C.G.
SMALL PROBE NO. 1 ALIGNMENT ACCURACY	10	14	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 2 ALIGNMENT ACCURACY	10	14	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 3 ALIGNMENT ACCURACY	16	16	±0.05 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SUBTOTAL WORST CASE	(880)	(917)	
SUBTOTAL RSS	(418)	(430)	
TOTAL WORST CASE	1033	1068	
TOTAL RSS	427	439	
LIMIT VALUE (THOR/DELTA USER'S GUIDE)	1560	1880	±0.020 RADIAN I_X= 107.4, I_Y= 90.5, I_Z= 87.1 SLUG-FT^2
AVAILABLE MARGIN	1133	1441	USING RSS VALUES

Table 6G-5. Thor/Delta Probe Spacecraft Static Balance Tolerance Allocations (Stowed Condition)

SOURCE	TOLERANCE (IN.-LB)	COMMENTS
BALANCE (MACHINE) ACCURACY	1.0	AVERAGE BASED ON CLOCKWISE AND COUNTERCLOCKWISE BALANCING
PROPELLANT WEIGHT AND C.G. (TANK 1) 14.3 POUNDS	2.1	ALIGNMENT OF TANKS (±0.05 INCH) AND PROPELLANT EQUAL LOADING ACCURACY (±0.1 POUND EACH TANK AT 20 INCH RADIUS)
PROPELLANT WEIGHT AND C.G. (TANK 2) 14.3 POUNDS	2.1	
PROPELLANT WEIGHT AND C.G. (TANK 3) 14.3 POUNDS	2.1	
SPACECRAFT CENTERLINE ALIGNMENT	4.2	±0.005 INCH ALIGNMENT REPEATABILITY (DRY SPACECRAFT)
POST-ALIGNMENT EQUIPMENT CHANGES	1.0	ALLOWANCE FOR EQUIPMENT CHANGES WITHOUT REBALANCING
SUBTOTAL WORSTCASE	(12.5)	
SUBTOTAL ROOT-SUM-SQUARE (RSS)	(5.7)	
LARGE PROBE C.G. ACCURACY	3.2	±0.010 INCH RADIAL C.G. UNCERTAINTY
LARGE PROBE ALIGNMENT ACCURACY	7.9	±0.025 INCH ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY	3.3	±0.01 INCH RADIAL C.G. UNCERTAINTY AND ±0.10 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY	3.3	±0.01 INCH RADIAL C.G. UNCERTAINTY AND ±0.10 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 3 WEIGHT AND C.G. ACCURACY	3.3	±0.01 INCH RADIAL C.G. UNCERTAINTY AND ±0.10 POUND WEIGHT UNCERTAINTY
SMALL PROBE NO. 1 ALIGNMENT ACCURACY	1.7	±0.03 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 2 ALIGNMENT ACCURACY	1.7	±0.03 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SMALL PROBE NO. 3 ALIGNMENT ACCURACY	1.7	±0.03 ALIGNMENT ACCURACY WITH RESPECT TO SPACECRAFT CENTERLINE
SUBTOTAL WORST CASE	(26.1)	
SUBTOTAL RSS	(10.7)	
TOTAL WORST CASE	38.6	
TOTAL RSS	12.1	
LIMIT VALUE (SPECIFICATION)	12.7	±0.015 INCH UNCERTAINTY IN 849-POUND SPACECRAFT
AVAILABLE MARGIN	0.6	USING RSS VALUES

Table 6G-6. Thor/Delta Probe Spacecraft Dynamic Balance Tolerance Allocations (Stowed Condition)

SOURCE	TOLERANCE (LB-IN.^{2}) P_{XY}	P_{XZ}	COMMENTS
BALANCE (MACHINE ACCURACY)	80	80	AVERAGE BASED ON CLOCKWISE AND COUNTERCLOCKWISE BALANCING
PROPELLANT WEIGHT AND C.G. (TANK 1)	14	22	±0.05 INCH UNCERTAINTY IN RADIAL AND LONGITUDINAL C.G. AND ±0.1 POUND PROPELLANT WEIGHT UNCERTAINTY IN EACH TANK
PROPELLANT WEIGHT AND C.G. (TANK 2)	14	22	
PROPELLANT WEIGHT AND C.G. (TANK 3)	25	7	
POST-ALIGNMENT EQUIPMENT CHANGES	20	20	
SUBTOTAL WORST CASE	(153)	(151)	
SUBTOTAL ROOT-SUM-SQUARE (RSS)	(88)	(88)	
LARGE PROBE PRINCIPAL AXIS	13	13	±13 LB-IN.^{2} UNCERTAINTY IN BALANCE OF PROBE
LARGE PROBE C.G. ACCURACY	55	55	±0.01 INCH RADIAL C.G.
LARGE PROBE ALIGNMENT ACCURACY	139	139	±0.025 INCH UNCERTAINTY IN ALIGNMENT RADIALLY
SMALL PROBES PRINCIPAL AXES	2	2	±1 LB-IN.^{2} EACH PROBE
SMALL PROBE NO. 1 WEIGHT AND C.G. ACCURACY	17	30	±0.10 POUND WEIGHT UNCERTAINTY AND ±0.1 INCH LONGITUDINAL C.G. AND ±0.01 INCH RADIAL C.G.
SMALL PROBE NO. 2 WEIGHT AND C.G. ACCURACY	17	30	±0.10 POUND WEIGHT UNCERTAINTY AND ±0.1 INCH LONGITUDINAL C.G. AND ±0.01 INCH RADIAL C.G.
SMALL PROBE NO. 3 WEIGHT AND C.G. ACCURACY	35	1	±0.10 POUND WEIGHT UNCERTAINTY AND ±0.1 INCH LONGITUDINAL C.G. AND ±0.01 INCH RADIAL C.G.
SMALL PROBE NO. 1 ALIGNMENT ACCURACY	6	8	±0.03 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 2 ALIGNMENT ACCURACY	6	8	±0.03 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SMALL PROBE NO. 3 ALIGNMENT ACCURACY	10	10	±0.03 INCH ALIGNMENT UNCERTAINTY LONGITUDINALLY AND RADIALLY
SUBTOTAL WORST CASE	(300)	(296)	
SUBTOTAL RSS	(157)	(157)	
TOTAL WORST CASE	453	447	
TOTAL RSS	180	180	
LIMIT VALUE (SPECIFICATION)	156	188	±0.002 RADIAN
AVAILABLE MARGIN	-24	8	USING RSS VALUES

APPENDIX 6H

DETAILED MASS PROPERTIES
OPTIONAL THOR/DELTA ORBITER CONFIGURATIONS
VERSION III SCIENCE PAYLOAD

APPENDIX 6H

DETAILED MASS PROPERTIES OPTIONAL THOR/DELTA ORBITER CONFIGURATIONS, VERSION III SCIENCE PAYLOAD

Thor/Delta Orbiter, 12-Watt Fanbeam, Fanscan Configuration, Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
ELECTRICAL POWER		30.4	66.9
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	11.07	24.4
BATTERY (16-12 AH NI-CD CELLS)	1	10.48	23.1
POWER CONTROL UNIT	1	6.35	14.0
DC-DC CONVERTER	1	2.45	5.4
COMMUNICATIONS		11.5	25.4
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
TRANSMITTER DRIVERS	2	1.09	2.4
POWER AMPLIFIERS	4	1.09	2.4
HYBRIDS	5	0.23	0.5
DIPLEXERS	2	1.95	4.3
SWITCHES	5	1.36	3.0
FANBEAM ANTENNA	1	1.13	2.5
FANSCAN ANTENNA	1	0.45	1.0
FORWARD OMNI ANTENNA	1	0.14	0.3
AFT OMNI ANTENNA	1	0.23	0.5
RF COAX AND CONNECTORS	AS REQUIRED	1.13	2.5
ELECTRICAL DISTRIBUTION		12.6	27.8
COMMAND DISTRIBUTION UNIT	1	3.54	7.8
HARNESS AND CONNECTORS	AS REQUIRED	9.07	20.0
DATA HANDLING		5.7	12.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	1.82	4.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		2.3	5.1
CONTROL ELECTRONICS ASSEMBLY	1	1.91	4.2
SUN SENSOR ASSEMBLY	2	0.41	0.9
PROPULSION		6.8	14.9
PROPELLANT TANK ASSEMBLY	3	3.13	6.9
THRUSTER ASSEMBLY	8	2.18	4.8
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATERS AND MISCELLANEOUS	AS REQUIRED	0.91	2.0

Thor/Delta Orbiter, 12-Watt Fanbeam, Fanscan Configuration
Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
SOLID INSERTION MOTOR (BURNOUT)	1	9.1	20.0
THERMAL CONTROL		11.2	24.8
INSULATION	AS REQUIRED	5.49	12.1
FORWARD CLOSURE	1	0.82	1.8
SIDE CLOSURE	1	0.41	0.9
LOUVER	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	0.91	2.0
STRUCTURE		41.9	92.3
CENTRAL CYLINDER ASSEMBLY		12.07	26.6
UPPER RING	1		(4.6)
CYLINDER	1		(8.8)
PLATFORM SUPPORT RING	1		(1.3)
SEPARATION RING	1		(7.9)
MOTOR MOUNTING RING	1		(3.2)
ATTACH HARDWARE	AS REQUIRED		(0.8)
PLATFORM/COMPARTMENT ASSEMBLY		13.88	30.6
UPPER STRUTS	15		(2.8)
LOWER STRUTS	9		(1.9)
VERTICALS	9		(2.3)
UPPER RING ASSEMBLY	1		(3.0)
PLATFORM STRUT FITTINGS	9		(1.4)
PLATFORM ASSEMBLY	1		(17.1)
MISCELLANEOUS BRACKETS AND ATTACH HARDWARE	AS REQUIRED		(2.1)
SOLAR ARRAY SUPPORT		3.67	8.1
UPPER RING	1		(2.6)
LOWER RING	1		(3.2)
STRUTS	18		(2.3)
ANTENNA SUPPORT ASSEMBLY	1	2.68	5.9
MAGNETOMETER BOOM ASSEMBLY	1	2.72	6.0
PROPULSION SUPPORT	AS REQUIRED	1.81	4.0
DAMPER	1	1.81	4.0
AFT OMNI SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	0.91	2.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	2.09	4.6

Thor/Delta Orbiter, 12-Watt Fanbeam, Fanscan Configuration
Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
BALANCE WEIGHT PROVISION		2.7	6.0
• SPACECRAFT BUS LESS SCIENCE		134.2	295.7
SCIENTIFIC INSTRUMENTS		28.3	62.5
MAGNETOMETER	1	2.27	5.0
ELECTRON TEMPERATURE PROBE	1	1.13	2.5
NEUTRAL MASS SPECTROMETER	1	4.54	10.0
ION MASS SPECTROMETER	1	1.36	3.0
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.08	9.0
RF ALTIMETER	1	9.53	21.0
• SPACECRAFT (DRY)		162.5	358.2
PROPELLANTS AND PRESSURANT		101.8	224.4
INSERTION PROPELLANT AND EXPENDED INERTS		84.41	186.1
HYDRAZINE PROPELLANT		17.10	37.7
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		264.3	582.6
CONTINGENCY (NET ALLOWABLE)		28.3	62.4
• GROSS SPACECRAFT AFTER SEPARATION		292.6	645.0

Thor/Delta Orbiter, Earth-Pointing Configuration, Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
ELECTRICAL POWER		30.4	66.9
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	11.07	24.4
BATTERY (16-12 AH NI-CD CELLS)	1	10.48	23.1
POWER CONTROL UNIT	1	6.35	14.0
DC-DC CONVERTER	1	2.45	5.4
COMMUNICATIONS		15.4	34.0
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
TRANSMITTER DRIVERS	2	1.09	2.4
POWER AMPLIFIERS	2	0.54	1.2
HYBRIDS	2	0.09	0.2
DIPLEXERS	2	1.95	4.3
SWITCHES	7	1.91	4.2
HIGH-GAIN ANTENNA ASSEMBLY	1	4.54	10.0
MEDIUM-GAIN ANTENNA	1	0.91	2.0
FORWARD OMNI ANTENNA	1	0.41	0.9
AFT OMNI ANTENNA	1	0.14	0.3
RF COAX AND CONNECTORS	AS REQUIRED	1.13	2.5
ELECTRICAL DISTRIBUTION		12.6	27.8
COMMAND DISTRIBUTION UNIT	1	3.54	7.8
HARNESS AND CONNECTORS	AS REQUIRED	9.07	20.0
DATA HANDLING		5.7	12.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	1.82	4.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		2.3	5.1
CONTROL ELECTRONICS ASSEMBLY	1	1.91	4.2
SUN SENSOR ASSEMBLY	1	0.41	0.9
PROPULSION		6.8	14.9
PROPELLANT TANK ASSEMBLY	3	3.13	6.9
THRUSTER ASSEMBLY	8	2.18	4.8
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATERS AND MISCELLANEOUS	AS REQUIRED	0.91	2.0

Thor/Delta Orbiter, Earth-Pointing Configuration,
Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
SOLID INSERTION MOTOR (BURNOUT)	1	9.1	20.0
THERMAL CONTROL		11.2	24.8
INSULATION	AS REQUIRED	5.49	12.1
FORWARD CLOSURE	1	0.82	1.8
SIDE CLOSURE	1	0.41	0.9
LOUVER	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	0.91	2.0
STRUCTURE		44.2	97.4
CENTRAL CYLINDER ASSEMBLY		12.07	26.6
UPPER RING	1		(4.6)
CYLINDER	1		(8.8)
PLATFORM SUPPORT RING	1		(1.3)
SEPARATION RING	1		(7.9)
MOTOR MOUNTING RING	1		(3.2)
ATTACH HARDWARE	AS REQUIRED		(0.8)
PLATFORM/COMPARTMENT ASSEMBLY		13.88	30.6
UPPER STRUTS	15		(2.8)
LOWER STRUTS	9		(1.9)
VERTICALS	9		(2.3)
UPPER RING ASSEMBLY	1		(3.0)
PLATFORM STRUT FITTINGS	9		(1.4)
PLATFORM ASSEMBLY	1		(17.1)
MISCELLANEOUS BRACKETS AND ATTACH HARDWARE	AS REQUIRED		(2.1)
SOLAR ARRAY SUPPORT		3.67	8.1
UPPER RING	1		(2.6)
LOWER RING	1		(3.2)
STRUTS	18		(2.3)
NM/IM SPECTROMETER BOOM ASSEMBLY	1	4.54	10.0
MAGNETOMETER BOOM ASSEMBLY	1	2.72	6.0
PROPULSION SUPPORT	AS REQUIRED	1.81	4.0
DAMPER	1	1.81	4.0
FORWARD OMNI SUPPORT	1	0.23	0.5
AFT OMNI SUPPORT	1	0.23	0.5
MEDIUM-GAIN SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	0.91	2.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	2.09	4.6

Thor/Delta Orbiter, Earth-Pointing Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
BALANCE WEIGHT PROVISION		2.7	6.0
• SPACECRAFT BUS LESS SCIENCE		140.4	309.4
SCIENTIFIC INSTRUMENTS		28.3	62.5
MAGNETOMETER	1	2.27	5.0
ELECTRON TEMPERATURE PROBE	1	1.13	2.5
NEUTRAL MASS SPECTROMETER	1	4.54	10.0
ION MASS SPECTROMETER	1	1.36	3.0
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.08	9.0
RF ALTIMETER	1	9.53	21.0
• SPACECRAFT (DRY)		168.7	371.9
PROPELLANTS AND PRESSURANT		101.8	224.4
INSERTION PROPELLANT AND EXPENDED INERTS		84.41	186.1
HYDRAZINE PROPELLANT		17.10	37.7
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		270.5	596.3
CONTINGENCY (NET ALLOWABLE)		22.1	48.7
• GROSS SPACECRAFT AFTER SEPARATION		292.6	645.0

Thor/Delta Orbiter, Despun Reflector Configuration, Version III Science Payload

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
ELECTRICAL POWER		34.7	76.4
SOLAR ARRAY (INCLUDING SUBSTRATE)	6 PANELS	12.34	27.2
BATTERY (16-15 AH NI-CD CELLS)	1	13.06	28.8
POWER CONTROL UNIT	1	6.35	14.0
DC-DC CONVERTER	1	2.90	6.4
COMMUNICATIONS		12.4	27.4
CONSCAN PROCESSOR	1	0.36	0.8
RECEIVERS	2	2.36	5.2
TRANSMITTER DRIVERS	2	1.09	2.4
POWER AMPLIFIERS	4	1.09	2.4
HYBRIDS	5	0.23	0.5
DIPLEXERS	2	1.95	4.3
SWITCHES	5	1.36	3.0
HIGH-GAIN ANTENNA ASSEMBLY	1	2.04	4.5
FANSCAN ANTENNA	1	0.45	1.0
FORWARD OMNI ANTENNA	1	0.14	0.3
AFT OMNI ANTENNA	1	0.23	0.5
RF COAX AND CONNECTORS	AS REQUIRED	1.13	2.5
ELECTRICAL DISTRIBUTION		12.6	27.8
COMMAND DISTRIBUTION UNIT	1	3.54	7.8
HARNESS AND CONNECTORS	AS REQUIRED	9.07	20.0
DATA HANDLING		5.7	12.5
DIGITAL TELEMETRY UNIT	1	3.08	6.8
DATA STORAGE UNIT	3	1.82	4.0
DIGITAL DECODER UNIT	2	0.77	1.7
ATTITUDE CONTROL		12.4	27.4
CONTROL ELECTRONICS ASSEMBLY	1	1.91	4.2
SUN SENSOR ASSEMBLY	2	0.41	0.9
DESPIN CONTROL ASSEMBLY	2	3.63	8.0
DESPIN DRIVE ASSEMBLY	1	6.49	14.3
PROPULSION		6.8	14.9
PROPELLANT TANK ASSEMBLY	3	3.13	6.9
THRUSTER ASSEMBLY	8	2.18	4.8
FILTER	1	0.18	0.4
PRESSURE TRANSDUCER	1	0.18	0.4
FILL AND DRAIN VALVE	1	0.18	0.4
LINE/HEATERS AND MISCELLANEOUS	AS REQUIRED	0.91	2.0

Thor/Delta Orbiter, Despun Reflector Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	(LB)
SOLID INSERTION MOTOR (BURNOUT)	1	9.1	20.0
THERMAL CONTROL		11.2	24.8
INSULATION	AS REQUIRED	5.49	12.1
FORWARD CLOSURE	1	0.82	1.8
SIDE CLOSURE	1	0.41	0.9
LOUVER	5 SQ FT	2.95	6.5
THERMAL FIN – TRANSMITTER	AS REQUIRED	0.68	1.5
HEATERS, ISOLATORS, PAINT, ETC.	AS REQUIRED	0.91	2.0
STRUCTURE		40.9	90.1
CENTRAL CYLINDER ASSEMBLY		12.07	26.6
UPPER RING	1		(4.6)
CYLINDER	1		(8.8)
PLATFORM SUPPORT RING	1		(1.3)
SEPARATION RING	1		(7.9)
MOTOR MOUNTING RING	1		(3.2)
ATTACH HARDWARE	AS REQUIRED		(0.8)
PLATFORM/COMPARTMENT ASSEMBLY		13.88	30.6
UPPER STRUTS	15		(2.8)
LOWER STRUTS	9		(1.9)
VERTICALS	9		(2.3)
UPPER RING ASSEMBLY	1		(3.0)
PLATFORM STRUT FITTINGS	9		(1.4)
PLATFORM ASSEMBLY	1		(17.1)
MISCELLANEOUS BRACKETS AND ATTACH HARDWARE	AS REQUIRED		(2.1)
SOLAR ARRAY SUPPORT		3.67	8.1
UPPER RING	1		(2.6)
LOWER RING	1		(3.2)
STRUTS	18		(2.3)
ANTENNA SUPPORT ASSEMBLY	1	1.45	3.2
MAGNETOMETER BOOM ASSEMBLY	1	2.72	6.0
PROPULSION SUPPORT	AS REQUIRED	1.81	4.0
DAMPER	1	1.81	4.0
FORWARD OMNI SUPPORT	1	0.23	0.5
AFT OMNI SUPPORT	1	0.23	0.5
SCIENCE SUPPORT BRACKETRY	AS REQUIRED	0.91	2.0
EQUIPMENT TIEDOWN AND INTEGRATED HARDWARE	AS REQUIRED	2.09	4.6

Thor/Delta Orbiter, Despun Reflector Configuration, Version III Science Payload (Continued)

DESCRIPTION	NUMBER REQUIRED	WEIGHT (KG)	WEIGHT (LB)
BALANCE WEIGHT PROVISION		2.7	6.0
• SPACECRAFT BUS LESS SCIENCE		148.5	327.3
SCIENTIFIC INSTRUMENTS		28.3	62.5
MAGNETOMETER	1	2.27	5.0
ELECTRON TEMPERATURE PROBE	1	1.13	2.5
NEUTRAL MASS SPECTROMETER	1	4.54	10.0
ION MASS SPECTROMETER	1	1.36	3.0
UV SPECTROMETER	1	5.44	12.0
IR RADIOMETER	1	4.08	9.0
RF ALTIMETER	1	9.53	21.0
• SPACECRAFT (DRY)		176.8	389.8
PROPELLANTS AND PRESSURANT		101.8	224.4
INSERTION PROPELLANT AND EXPENDED INERTS		84.41	186.1
HYDRAZINE PROPELLANT		17.10	37.7
NITROGEN PRESSURANT		0.27	0.6
• SPACECRAFT LESS CONTINGENCY		278.6	614.2
CONTINGENCY (NET ALLOWABLE)		14.0	30.8
• GROSS SPACECRAFT AFTER SEPARATION		292.6	645.0

APPENDIX 6I

FAILURE MODE AND EFFECTS ANALYSIS

APPENDIX 6I
FAILURE MODE AND EFFECTS ANALYSIS

The following tables are the failure mode and effects analyses for the Thor/Delta and Atlas/Centaur large and small probes. These tables only reflect singular failures/primary failure effects and indicate the associated functions, and failure modes. These failure mode and effects analyses were used to identify critical equipment and high-risk functions as subjects for system configuration trades and reliability versus resource allocation studies (weight and volume).

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis

Subsystem/Item	Function	Failure Mode	Failure effect
I. Electrical Power/Ordnance			
A. Battery	Supply probe electrical energy	Cell open	Catastrophic. Total loss of all LP Mission objectives due to loss of power.
		Cell short	Loss of margin. Cells are sized with margin such that Battery has cell out (short) capability.
B. Battery Heater	Bring Battery up to optimum operating temperature for entry.	Open	Major Degradation. Battery energy compromised with rated capacity unobtainable.
C. Power Control Unit			
o Power Transfer Relay	Power Up Probe Bus by bringing battery on line.	Fail to make	None. Dual relays, redundant in make mode.
		Fail to break	Not applicable to entry-descent period; only after checkout. Probe Bus would remain powered up with battery on line.
o Safe/Arm Relay	Safe/Arm the various probe ordnance functions.	Fail to make	Critical. No ordnance functions due to pyro fire control not powered - Drogue Mortar, Base Cover Separation, Aeroshell Separation, Main Chute Release, Mass Spectrometer Inlet Tubes.
		Fail to break	None. Requires a secondary (erroneous signals) failure to create a problem.
o Pyro Fire Control	Triggering energy for the various probe ordnance functions.	Open/short	None. Redundant, isolated, and separate triggering circuits for the redundant, isolated, and separate ordnance buses.
o Power Switches	Power the various probe experiments.	Fail to make	Compromise LP mission due to loss of individual experiments.
		Fail to break	Only applicable after checkout and just prior to impact; individual experiment power drain.
o Window Heater Controls	Prevent scientific window condensation/distortion due to inner surface/outer surface delta temperatures	Fail open	Degrade individual experiments if window condensation/distortion occurs.

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
o Fuses	Protect Probe Bus from individual experiment shorts.	fail open	Minor degradation. Individual experiment loss.
D. Chute Mortar	Deploy chute	Fail inoperative	Critical. Chute not deployed, therefore Aeroshell Forebody will not separate and probe remains buttoned up. Note: redundant initiators/redundant pyro buses.
E. Nut Separator Ordnance Devices for Aeroshell Separation (3 devices)	Retain/Release Aeroshell Forebody.	Fail inoperative	Critical. Aeroshell Forebody not separated and probe remains buttoned up. Note: redundant initiators/redundant pyro buses.
F. Chute Release Pin Pullers (3 devices)	Retain/Release Chute	Fail inoperative	Major degradation. Probe remains on chute and base cover retained with battery exhaustion/overheating before lower altitude data completely obtained. Note: redundant initiators/redundant pyro buses.
G. Interface Cable Cutter (separation)	Sever Bus/Probe umbilical. TRW side of interface.	Fail inoperative	Catastrophic. Total loss of all LP mission objectives due to preclusion of probe separation.
H. Aeroshell Forebody Electrical Cable Cutter	Sever Probe Aeroshell Forebody/ Base Cover Descent Capsule umbilical to permit Aeroshell Forebody separation.	Fail inoperative	Critical. Aeroshell Forebody doesn't separate and probe remains buttoned up. Note: redundant initiators/redundant pyro buses.
I. Base Cover Electrical Cable Cutter	Sever Probe Aeroshell Base Cover/Descent Capsule umbilical to permit Descent Capsule separation	Fail inoperative	Major degradation. Probe remains on chute due to Base Cover retention with battery exhaustion/probe overheating before lower altitude data completely obtained. Note: redundant initiators/redundant pyro buses.
II. Data Handling/Command			
A. Timing Generator			
o Oscillator & Count-down Circuit	Provide system clocks	Inoperative or locked up	None or no further ordnance events; no format changes; countdown failure with loss of LP mission due to no more data.
oo Frame Start Gating	Frame rate control	Locked in one position - Locked in other position	Loss of LP mission due to loss of data. Serious degradation of LP data due to data module loss.

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
o Bit-Rate Counter	Control Data Rate	Inoperative	Serious degradation of LP data due to data modulation loss.
o Word-Rate Counter	Control Bit/Word	Inoperative	Serious degradation of LP data due to data modulation loss.
o Frame-Rate Counter	Control Words/Frame	Inoperative	Serious degradation of LP data due to data modulation loss.
o Descent timer	Initiate descent discrete events	Inoperative	Serious degradation of LP mission due to loss of terminal descent data, further ordnance events, no format changes, no bit-rate changes, etc.
o Line Select	Selects internal or external command mode	Locked in Bus Command Mode	Loss of LP mission due to loss of internal sequencing.
		Loss of Bus Cmd	Loss of capability for preseparation checkout
		Inoperative	Incorrect or no sequence for LP mission loss
o PROM & Sequencer	Decode event times and supply discrete event signals.	Locked up	Serious degradation of LP data due to loss of discrete events; ordnance, bit rate changes, etc.
o "g" switches	Detect Probe atmosphere entry; back up 25 day cruise timer and reset the descent timer.	Fail inoperative	None. Dual "g" switches, redundant
o Coast Timer	Turn on IR & Battery Heaters and start entry sequence.	Fail inoperative	Serious degradation. Probe not turned on at entry results in loss of high altitude data; but entry sequence startup back up by "g" switches assures low altitude
			Turn on (descent) but compromised due to loss of IR chamber heater.
B. Decoder	Decodes serial command word	Fail inoperative prior to, or at preseparation checkout.	Possibly negate completion of preseparation checkout.
C. Format Generator			
o PROM D_1	Select channel to be sampled	Fail inoperative	Critical. Loss of LP mission data from 70 km to 42 km.
o PROM D_2		Fail inoperative	Critical. Take over D_1 function at 42 km altitude. Lose low altitude data.[1]
o PROM A		Fail inoperative	Serious degradation of LP mission data
o PROM B		Fail inoperative	Serious degradation of LP mission data

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
C. Format Generator (Cont.)			
o Decoder/Selector	Decode 8-bit PROM output to 1 of 256 channels	Fail inoperative	Complete loss or degradation of LP mission data.
D. Multiplexer			
o Bilevel	Sample bi-level data	Fail inoperative	Serious LP mission degradation due to loss of bi-level data.
o Single ended, high level	Sample high level analog channels	Fail inoperative	Serious LP mission degradation due to loss of single ended, high level data.
o Differential	Sample low level analog channels	Fail inoperative	LP mission degradation due to loss of Engineering data.
E. Data Combiner	Mix analog, digital, bi-level and stored data into bit stream.	Fail inoperative	Degradation of LP data modulation due to no, or mixed data output.
F. Signal Conditioning			
o Differential to single ended amplifier	Converts differential signal to single ended.	Fail inoperative	LP mission degradation due to loss of engineering data.
o Impedance Buffer	Eliminate loading of science signal.	Fail inoperative	Serious LP mission degradation due to loss of analog data.
o Bilevel Comparator	Determines one or zero signal	Fail inoperative	Serious LP mission degradation due to loss of bilevel data.
G. A/D Converter			
o A/D Converter	Converts Analog signal to digital word.	Fail inoperative	Loss of all analog data seriously degrading LP mission.
o Reference power supply	Provide reference for A/D converter.	Fail inoperative	Loss of all analog data seriously degrading LP mission.
H. Data Storage Unit			
o Small buffer memory (3 units)	Temporary outgoing real time data storage.	Fail inoperative	Degradation to LP mission due to loss of 1/3 mission data per buffer.
o Gating	Stored data transmission selection	Fail in one position	Loss of mission due to inability to store data
		Fail in other position	Loss of LP mission due to inability to transmit

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
H. Data Storage Unit (Cont)			
o Large Memory	Stored data storage	Fail inoperative	Minor degradation. Loss of stored data but real time data still available through small buffer memories.
I. Biphase Modulator and Convolutional Encoder	Reduce bit error rate by encoding/modulating data.	Fail short or open	Serious degradation of LP data modulation.
J. Command and Data Registers	Parallel data to serial bit stream conversion.	Fail inoperative prior to, or at preseparation checkout.	Possibly negate completion of preseparation checkout data readouts
K. DC/DC Converter	Converts 28 Vdc power to multiple, regulated power.	Fail inoperative (+12, -12, +5 Vdc)	Loss of LP mission due to loss of data.
L. Engineering Transducers			
o Transducer Completion Networks	Provide engineering measurements.	Fail inoperative	Minor degradation. Loss of engineering measurements.
III. Communication			
A. Phase Modulator & Transmitter Driver	Carrier modulation and power amplifier driver	Fail inoperative	Complete loss of LP mission due to loss of data.
B. S-Band Power Amplifier	Modulated signal amplification.	Fail inoperative	Complete loss of LP mission due to loss of data.
C. Diplexer	Transmitter/receiver isolation	Fail open or short	Complete loss or serious degradation of LP mission due to loss of data or loss of two-way doppler.
D. Receiver	Uplink signal receiver/ phase tracking	Noisy VCO	Serious degradation of mission due to loss of two-way doppler and/or telemetry.
E. Antenna	Signal transducer receive/ transmit	Fail inoperative	Complete loss of LP mission due to loss of data.
IV. Structure/Mechanisms			
A. Aeroshell, Fwd.	Provide structural shell	Fail Thermo-structurally	Complete loss or serious degradation of LP mission due to heat damage transferred to equipments and/or loss of stability due to collapse.

Table 6I-1. Thor/Delta Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
B. Aeroshell, Aft.	Provide structural shell	Fail Thermo-Structurally	Complete loss or serious degradation of LP mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
C. Heatshield	Provide aero-thermal Protection	Fail Thermo-Structurally	Complete loss or serious degradation of LP mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
D. Pressure Vessel	Provide Science and Support Equipments pressure protection.	Fail Thermo-Structurally	Complete loss or serious degradation of LP mission due to equipment damage by collapse.
E. Chute		Fail to Open	Critical. Loss of deceleration function, Aeroshell Forebody not separated and probe remains buttoned up.
o Swivel	Deceleration, Aeroshell	Fail Thermo-Structurally	
o Bridle	Forebody separation force, base cover removal		

Table 61-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis

Subsystem/Item	Function	Failure Mode	Failure Effect
I. Electrical Power/Ordnance			
A. Battery	Supply probe electrical energy	Cell open	Catastropic. Total loss of all SP Mission objectives due to loss of power.
		Cell short	Loss of margin. Cells are sized with margin such that battery has cell out (short) capability.
B. Battery Heater	Bring battery up to optimum operating temperature for entry.	Open	Major Degradation. Battery energy compromised with rated capacity obtainable.
C. Power Control Unit			
• Power Transfer Relay	Power Up Probe Bus by bringing battery on line.	Fail to make	None. Dual relays, redundant in make mode.
		Fail to break	Not applicable to entry-descent period, only after checkout. Probe Bus would remain powered up with battery on line.
• Safe/Arm Relay	Safe/Arm the two nonexplosive ordnance functions.	Fail to make	Minor degradation. Loss of temperature probe and nephelometer.
		Fail to break	None. Requires a secondary (erroneous signals) failure to create a problem.
• Fire Control-Nonexplosive Pinpullers.	Triggering energy for the two probe nonexplosive ordnance functions.	Fail inoperative by function.	
	Temperature Probe Deployment		Minor degradation. Loss of temperature probe.
	Nephelometer Window Cover Deployment		Minor degradation. Loss of nephelometer readings.
• Window Heater Control	Prevent scientific window condensation/distortion due to inner surface/outer surface delta temperature.	Fail open	Degrade individual experiments if window condensation/distortion occurs.

Table 6I-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
• Power Switches	Power the various probe experiments.	Fail to make	Compromise SP mission due to loss of individual experiments.
		Fail to break	Only applicable after checkout and just prior to impact; individual experiment power drain.
• Fuses	Protect Probe bus from individual experiment shorts.	Fail open	Minor degradation. Individual experiment loss.
D. Temperature Probe Deployment Device.	Deploy temperature probe	Fail inoperative	Minor degradation. Loss of temperature probe.
E. Nephelometer Window Cover Deployment Device.	Uncover nephelometer window	Fail inoperative	Minor degradation. Loss of nephelometer readings.
F. Interface Cable Cutter (separation).	Sever Bus/Probe umbilical. TRW side of interface.	Fail inoperative	Catastrophic. Total loss of all SP Mission objectives due to preclusion of probe separation.
II. Data Handling/Command (Thor/Delta only).			
A. Timing Generator			
• Oscillator and Countdown Circuit.	Provide system clocks	Inoperative or locked up.	None or no further nonexplosive pinpuller events; no format changes; countdown failure with loss of SP Mission due to no more data.
•• Frame Start Gating	Frame rate control	Locked in one position.	Loss of SP Mission due to loss of data.
		Locked in other position.	Serious degradation of SP data due to data modulation loss.
• Bit-Rate Counter	Control data rate	Inoperative	Serious degradation of SP data due to data modulation loss.
• Word-Rate Counter	Control bits/word	Inoperative	Serious degradation of SP data due to data modulation loss.
• Frame-Rate Counter	Control words/frame	Inoperative	Serious degradation of SP data due to data modulation loss.
• Descent Timer	Initiate descent discrete events	Inoperative	Serious degradation of SP Mission due to loss of terminal descent data, further nonexplosive pinpuller events, no format changes, no bit-rate changes, etc.

Table 6I-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
A. Timing Generator (cont.)			
• Line Select	Selects internal or external command mode.	Locked in Bus Command Mode	Loss of SP Mission due to loss of internal sequencing.
		Loss of Bus Command	Loss of capability for preseparation checkout.
		Inoperative	Incorrect or no sequence for SP Mission loss.
• PROM and Sequencer	Decode event times and supply discrete event signals.	Locked up	Serious degradation of SP data due to loss of discrete events; nonexplosive pinpuller, bit rate changes, etc.
• "g" switches	Detect probe atmospheric entry; backup 25 day cruise timer and reset the descent timer.	Fail inoperative	None. Dual "g" switches, redundant.
• Coast Timer	Turn on IR and Battery Heaters and start entry sequence.	Fail inoperative	Serious degradation. Probe not turned on at entry results in loss of high altitude data; but back up by "g" switches assures low altitude turn on (descent) but compromised due to loss of IR chamber heater.
B. Decoder	Decode serial command word.	Fail inoperative prior to, or at preseparation checkout.	Possibly negate completion of preseparation checkout.
C. Format Generator			
• PROM D	Select channel to be sampled.	Fail inoperative	90% loss of SP Mission data
• PROM A		Fail inoperative	Serious degradation of SP Mission data
• PROM B		Fail inoperative	Serious degradation of SP Mission data
• Decoder/Selector	Decode 8 bit PROM output to 1 of 256 channels.	Fail inoperative	Complete loss or degradation of SP Mission data
D. Multiplexer			
• Bilevel	Sample bilevel data	Fail inoperative	Serious SP Mission degradation due to loss of bilevel data.

Table 6I-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
D. Multiplexer (cont.)			
• Single ended, high level	Sample high level analog channels	Fail inoperative	Serious SP Mission degradation due to loss of single ended, high level data.
• Differential	Sample low level analog channels	Fail inoperative	Serious SP Mission degradation due to loss of engineering data.
E. Data Combiner	Mix analog, digital, bilevel and stored data into bit stream.	Fail inoperative	Degradation of SP data modulation due to no, or mixed, data output.
F. Signal Conditioning			
• Differential to single ended amplifier.	Convert differential signal to single ended	Fail inoperative	Serious SP Mission degradation due to loss of engineering data.
• Impedance Buffer	Eliminate loading of science signal.	Fail inoperative	Serious SP Mission degradation due to loss of analog data.
• Bilevel Comparator	Determine one or zero signal	Fail inoperative	Serious SP Mission degradation due to loss of bilevel data.
G. A/D Converter			
• A/D Converter	Convert analog signal to digital word.	Fail inoperative	Loss of all analog data seriously degrading SP Mission.
• Reference Power Supply	Provide reference for A/D converter.	Fail inoperative	Loss of all analog data seriously degrading SP Mission.
H. Data Storage Unit			
• Small Buffer Memory (3 units)	Temporary outgoing data storage	Fail inoperative	Degradation to SP Mission due to loss of 1/3 mission data per buffer.
• Gating	Stored data transmission selection	Fail in one position. Fail in other position.	Loss of SP Mission due to inability to store data. Loss of SP Mission due to inability to transmit data.
• Large Memory	Stored data storage	Fail inoperative	Minor degradation. Loss of stored data but real time data still available through small buffer memories.

Table 6I-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
I. Biphase Modulator and Convolutional Encoder.	Reduce bit-error rate by encoding/modulating data.	Fail short or open	Serious degradation of SP data modulation
J. Command and Data Registers	Parallel data to serial bit stream conversion.	Fail inoperative prior to, or at preseparation, checkout.	Possibly negate completion of preseparation checkout data readouts.
K. DC/DC Converter	Converts 28 Vdc power to multiple, regulated power.	Fail inoperative (+12, -12, +65 Vdc)	Loss of SP Mission due to loss of data.
L. Engineering Transducers	Provide engineering measurements	Fail inoperative	Minor degradation. Loss of engineering measurements.
• Transducer Completion Networks.			
III. Data Handling/Command (Atlas/Centaur only)			
• Data Transmission Unit	Data Handling/Command	Fail inoperative	Critical degradation of SP Mission due to loss of all data except modulation.
IV. Communication			
A. Phase Modulator and Transmitter Driver.	Carrier modulation and power amplifier driver.	Fail inoperative	Complete loss of SP Mission due to loss of data.
B. S-band Power Amplifier	Modulated signal amplification	Fail inoperative	Complete loss of SP Mission due to loss of data.
C. Antenna	Signal transducer receiver/transmit.	Fail inoperative	Complete loss of SP Mission due to loss of data.
V. Structure/Mechanisms			
A. Aeroshell, Fwd.	Provide structural shell	Fail Thermo-Structurally	Complete loss or serious degradation of SP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.

Table 6I-2. Thor/Delta and Atlas/Centaur Small Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
B. Aeroshell, Aft.	Provide structural shell	Fail Thermo-Structurally	Complete loss or serious degradation of SP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
C. Heat Shield	Provide aero-thermal protection	Fail Thermo-Structurally	Complete loss or serious degradation of SP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
D. Pressure Vessel	Provide science and support equipments pressure protection.	Fail Thermo-Structurally	Complete loss or serious degradation of SP Mission due to equipment damage by collapse.

Table 6I-3. Atlas/Centaur Large Probe Failure Mode and Effects Analysis

Subsystem/Item	Function	Failure Mode	Failure Effect
I. Electrical Power/Ordnance			
A. Battery (two required for total mission)	Supply probe electrical energy	Cell open	50% energy loss. Loss of one of two batteries.
		Cell short	Loss of one battery's margin. Cells are sized with margin such that battery has cell out (short) capability.
B. Battery Heater (two)	Brings battery up to optimum operating temperature for entry.	Open	Possible 50% energy loss. Battery energy compromised with rated capacity unobtainable.
C. Battery Isolation Diode (two)	Protect one battery from the other battery's failure.	Fail open	50% energy loss. Loss of one of two batteries.
		Fail short	Loss of failure protection.
D. Power Control Unit			
• Power Transfer Relay	Power up probe bus by bringing battery on line.	Fail to Make	None. Dual relays, redundant in make mode.
		Fail to Break	Not applicable to entry-descent period; only after checkout. Probe bus would remain powered up with battery on line.
• Safe/Arm Relay	Safe/Arm the various probe ordnance functions.	Fail to Make	Critical. No ordnance functions due to pyro fire control not powered - Pilot Mortar, Aeroshell separation, Afterbody/Main chute separation, Mass Spectrometer Inlet Tubes.
		Fail to Break	None. Requires a secondary (erroneous signals) failure to create a problem.
• Pyro Fire Control	Triggering energy for the various probe ordnance functions.	Open/Short	None. Redundant, isolated, and separate triggering circuits for the redundant, isolated, and separate ordnance buses.
• Power Switches	Power the various probe experiments.	Fail to Make	Compromise LP mission due to loss of individual experiments.

Table 6I-3. Atlas/Centaur Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
• Window Heater Controls	Prevent scientific window condensation/distortion due to inner surface/outer surface delta temperatures.	Fail open	Degrade individual experiments if window condensation/distortion occurs.
• Fuses	Protect probe bus from individual experiment shorts.	Fail open	Minor degradation. Individual experiment loss.
E. Pilot Chute Mortar	Deploy pilot chute	Fail inoperative	Critical. Pilot chute note deployed, aft thermal cover not removed, main chute not deployed, aeroshell forebody wont separate and probe remains buttoned up. Note: redundant initiators/redundant pyro buses.
F. Not Separator Ordnance Devices for Aeroshell Separation (3 devices)	Retain/Release aeroshell forebody.	Fail inoperative	Critical. Aeroshell forebody not separated and probe remains buttoned up. Note: redundant initiators/redundant pyro buses.
G. Not Separator Ordnance Devices for Afterbody/ Main Chute Separation.	Retain/Release afterbody/ main chute.	Fail inoperative	Major degradation. Probe remains on chute due to base cover retention with battery exhaustion/probe overheating before lower altitude data completely obtained. Note: redundant initiators/redundant pyro buses.
H. Interface Cable Cutter (separation)	Sever bus/probe umbilical. TRW side of interface.	Fail inoperative	Catastrophic. Total loss of all LP Mission objectives due to preclusion of probe separation.
I. Staging Connector (Afterbody/ Descent Capsule).	Provide connection between pyro/umbilical wires between afterbody and descent capsule.	Fail to disconnect	None. No locking mechanism, both sides of connector are held by structures with spring assist separation. Differential separation forces of afterbody/chute and descent capsule backs up the spring assists.
II. Data Handling/Command			
A. Data Transmission Unit	Data Handling/Command	Fail inoperative	Serious degradation of LP mission due to loss of all data except modulation.

Table 6I-3. Atlas/Centaur Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
III. Communication			
A. Transponder			
• Modulator-Driver	Carrier Modulation and Power Amplifier Driver	Fail inoperative	Complete loss of LP Mission due to loss of data.
• Receiver	Uplink Signal Receiving/Phase Tracking.	Fail inoperative	Serious degradation of mission due to loss of two way Doppler and/or telemetry.
B. S-band Power Amplifiers (two)	Modulated signal amplification	Fail inoperative	Complete loss of LP Mission due to loss of data.
• Hybrid Couplers (two)	Couple amplifier signals	Fail inoperative	Complete loss of LP Mission due to loss of data.
C. Diplexer	Transmitter/receiver isolation	Fail open or short	Complete loss or serious degradation of LP Mission due to loss of data or loss of two way Doppler.
D. Antenna	Signal transducer receive/transmit	Fail inoperative	Complete loss of LP Mission due to loss of data.
IV. Structure/Mechanisms			
A. Aeroshell, Fwd	Provide structural shell	Fail Thermo-Structurally	Complete loss or serious degradation of LP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
B. Afterbody	Provide structural shell	Fail Thermo-Structurally	Complete loss or serious degradation of LP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
C. Heatshield	Provide aero-thermal protection	Fail Thermo-Structurally	Complete loss or serious degradation of LP Mission due to heat damage transferred to equipments and/or loss of stability due to collapse.
D. Pressure Vessel	Provide science and support equipments pressure protection.	Fail Thermo-Structurally	Complete loss or serious degradation of LP mission due to equipment damage by collapse.

Table 6I-3. Atlas/Centaur Large Probe Failure Mode and Effects Analysis (Continued)

Subsystem/Item	Function	Failure Mode	Failure Effect
E. Pilot Chute • Riser • Bridle • Extraction Bridle	Remove aft thermal cover, extract main chute deployment bag and in turn deploy main chute.	Fail to Open Fail Thermo-Structurally	Critical. Loss of deceleration function, aeroshell forebody not separated and probe remains buttoned up.
F. Main Chute • Swivel • Bridle	Deceleration Aeroshell forebody separation force.	Fails to open Fail Thermo-Structurally	Critical. Loss of deceleration function, aeroshell forebody not separated and probe remains buttoned up.

Table 6I-4. System Failure Mode and Effects Analysis

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM ELECTRICAL POWER

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
1-1	Solar Array	Supply electrical power to spacecraft	Loss of strings in solar array/open cell or open solder joints	Loss of portion of redundancy.	None	60 string array with only 57 strings required.
1-2	Power Control Unit	Provide 28 volt power distribution.	Loss of 28 volt power distribution.	Loss of redundancy.	Telemetry	Internally redundant.
		Array control through shunt regulator	Loss of solar array power control.	Loss of redundancy.	Telemetry	Internally redundant.
		Battery charge regulation.	Loss of battery charge capability.	Battery will not continually discharge.	Telemetry	Battery is only required for first hour and pulse loads there alter so circuit is only supplied for backup to other failures.
1-3	Shunt Radiator	Provide for power loss control for solar array efficiency.	Loss of part of resistance networks in shunt radiator.	Loss of redundancy.	None	Shunt radiator made with redundant resistive elements.
1-4	Battery	Provide power in first hour of mission plus during pulse loads.	Short or open of a cell.	Loss of spacecraft in first hour, thereafter would affect the power loads.	Telemetry	None
1-5	DC-DC Converter	Provide secondary power.	Loss of converter.	Loss of redundancy.	Telemetry	Redundant converter.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM COMMUNICATION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
2-1	Forward Omni	Provide forward hemisphere command access to spacecraft.	Loss of antenna/ fracture of antenna or unit.	None. Nominal mission does not require forward hemisphere access.	Loss of spacecraft response.	Considered to be a structural item since passive and designed with margin of safety
2-2	Aft. Omni	Provide command access during maneuvers.	Loss of antenna/ fracture of antenna or unit.	Would require that maneuvers be performed blind with stored commands.	Loss of spacecraft performance.	Considered to be a structural item since passive and designed with margin of safety
2-3	Diplexer	Provide transmitter and receiver signal separation.	Crack or fracture during launch environment.	Loss of antenna.	Loss of spacecraft performance or spacecraft.	Considered to be a structural item since passive and carries no structural loads.
2-4	Transfer Switch	See Item 2-8				
2-5	Receiver	Provide command access to the spacecraft.	Loss of receiver due to internal failures.	Loss of redundancy.	Telemetry	Active redundant receivers.
2-6	Medium gain Antenna	Provide downlink for science data.	Loss of antenna/ failure of antenna mount.	Loss of probe bus science data.	Telemetry	Antenna is a passive device not meant as a structural item and is designed with a factor of safety.
2-7	Transmitter Amplifier	Provide downlink power.	Loss of amplifier due to internal failure.	Loss of redundancy.	Telemetry	Standby redundant downlink amplifiers.
2-8	Transfer Switch	Provide receiver and transmitter switching between antennas.	Failure to transfer.	Partial or complete loss of mission depending on when failure occuring.	Telemetry	The transfer switches are used only a few times and have high cyclic reliability.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA

SUBSYSTEM COMMUNICATION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
2-9	Hybrid	Provide power switching between RF components.	Loss of Hybrid.	Loss of Transmitter.	Telemetry	Hybrid is an inherently reliable device since passive and has a highly reliable history.
2-10	Transmitter	Provide downlink RF signal.	Loss of Transmitter.	Loss of Redundancy.	Telemetry.	Redundant transmitters.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM DATA HANDLING

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
3-1	Digital Decoding Unit	Provide for decoding of uplink.	Loss of digital decoding due to internal failure.	Loss of redundancy.	Telemetry	Active redundancy employed.
3-2	Digital Telemetry Unit	Provide telemetry data for downlink.	Loss of Digital Decoder unit due to internal failure.	Loss of redundancy.	Telemetry	Standby redundancy employed.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM ELECTRICAL DISTRIBUTION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
4-1	Command Distribution Unit	Provide command distribution and ordnance firing.	Loss of command distribution unit due to internal failure.	Loss of redundancy.	Telemetry	Active redundancy employed.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM ATTITUDE CONTROL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
5-1	Sensor and Power Control Unit	Provide attitude sensing and power control to attitude control components (black boxes).	Loss of sensor and power control due to piece part or internal failure.	Loss of redundancy.	Telemetry	Internal active redundancy.
5-2	Duration time Steering Logic	Provide logic for attitude control maneuvers.	Loss of duration and steering logic due to piece part failure or internal failure.	Loss of redundancy.	Telemetry	Standby redundancy employed.
5-3	Program Storage and Execute Unit	Provide attitude control command storage for maneuvers and execute commands.	Loss of program storage and execute unit due to piece part failure or internal failure.	Loss of onboard attitude control command capability.	Telemetry	Maneuver performed by ground command. Unit is required for only 50 hours.
5-4	Sun Sensor Electronics	Provide sun angle determination.	Loss of Sun Sensor Electronics due to piece part failure or internal failure.	Loss of redundancy.	Telemetry	Active redundancy.
5-5	Sun Sensor	Provide inputs to Sun Sensor for sun angle determination.	Loss of input of Sun Sensor electronics.	Loss of redundnacy.	Telemetry	Active redundancy.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM PROPULSION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
7-1	Pressure Transducers	Provide propellant useage data.	Fail to function or	Loss of direct measurement of propellant available.	Telemetry	Propellant determined through maneuvering rates.
			Leakage	Loss of propulsion subsystem (depending on leak rate).	None	Testing and quality control during installation.
7-2	Tanks	Provide for propellant storage.	Burst	Loss of propellant.	Telemetry	Burst test on tank and designed with margin of safety.
7-3	Fill and Drain Valve	Provide for filling and draining propellant tanks.	Leakage	Loss of hydrazine resulting in shorter life.	Telemetry	Redundant seals.
7-4	Thrusters	Provide impulse for maneuvers.	Leakage	Loss of propellant.	Telemetry	Redundant seals in thruster.
			Degraded catalyst bead.	Loss of ISP	Telemetry	Redundant thrusters.
7-5	Connectors and Manifold	Provide for propellant distribution.	Leakage at joint.	Loss of propellant.	Telemetry	All brassed connections and designed with margin of safety

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM STRUCTURE/THERMAL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
8-1	Structure	Provide platform for all other hardware.	Bending, buckling, fracture, fatigue.	Dependent upon structural element which fails.	May or may not be detectable by telemetry.	All structure designed with a factor of safety and margin of safety above 1.0.
8-2	Louvers	Provide for thermal control.	Failure of a louver to work.	Loss of redundancy in louver subsystem.	Telemetry	Designed so that loss of a thermal louver will not degrade thermal control.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA   SUBSYSTEM DEPLOYMENT

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
9-1	Small Probe Pin Pullers (2)	Release Probe	Failure of pin pullers to extract; due to insufficient gas.	Small probe not released and would unbalance the spacecraft.	Telemetry	Redundant firing circuits.
	Cable Cutter (1)	Cut umbilical.	Failure of cable cutter to cut umbilical.	Small probe not released and would unbalance the spacecraft.	Telemetry	Redundant firing circuits.
9-2	Large Probe Ball Locks (3)	Release large probe.	Failure of ball lock to operate; insufficient gas.	Large probe not released; may unbalance attitude control.	Telemetry	Redundant firing circuits.
	Cable Cutter (1)	Cut umbilical.	Failure of cable cutter to operate; insufficient gas.	Large probe not released; may unbalance attitude control.	Telemetry	Redundant firing circuits.
9-3	Other Release Mechanisms	Release other required equipment (magnatron for boom, etc.)	Failure of pin puller to extract due to insufficient gas.	Failure of experiment to deploy; may unbalance attitude control.	Telemetry	Redundant firing circuits.

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR SUBSYSTEM ELECTRICAL POWER

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
1-1	Solar Array	See 1-1 of Probe Bus/Thor Delta				
1-2	Power Control Unit	See 1-2 of Probe Bus/Thor Delta				
1-3	Shunt Radiator	See 1-3 of Probe Bus/Thor Delta				
1-4	Battery	See 1-4 of Probe Bus/Thor Delta				
1-5	Inverter	Provide power to central transformer rectifier (CTRF)	Loss of inverter due to piece part failure	Loss of redundancy.	Telemetry	Active redundant inverter
1-6	CTRF	Provide secondary power to individual components	Loss of CTRF	Loss of redundancy	Telemetry	Active redundancy with each component receiving isolated redundant secondary power

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR SUBSYSTEM COMMUNICATIONS

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
2-1	Forward Omni	See 2-1 of Thor/Delta				
2-2	Aft Omni	See 2-2 of Thor/Delta				
2-3	Diplexer	See 2-3 of Thor/Delta				
2-4	Transfer Switch	See 2-8 of Thor/Delta				
2-5	Receiver	See 2-5 of Thor/Delta				
2-6	Medium Gain Antenna	See 2-6 of Thor/Delta				
2-7	Transmitter Amplifier	See 2-7 of Thor/Delta				
2-8	Transfer Switch	See 2-8 of Thor/Delta				
2-9	Hybrid	See 2-9 of Thor/Delta				
2-10	Transmitter	See 2-10 of Thor/Delta				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR SUBSYSTEM DATA HANDLING

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
3-1	Digital Decoder Unit	See 3-1 of Thor/Delta				
3-2	Digital Telemetry Unit	See 3-2 of Thor/Delta				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR

SUBSYSTEM ELECTRICAL DISTRIBUTION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
4-1	Command Distribution Unit	See 4-1 of Thor/Delt				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR SUBSYSTEM ATTITUDE CONTROL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
5-1	Sensor and Power Control Unit	See 5-1 of Thor/Delt				
5-2	Duration Time Steering Logic	See 5-2 of Thor/Delt				
5-3	Program Storage and Execute Unit	See 5-3 of Thor/Delt				
5-4	Sun Sensor Electronics	See 5-4 of Thor/Delt				
5-6	Sun Sensor	See 5-5 of Thor/Delt				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR   SUBSYSTEM PROPULSION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
7-1	Pressure Transducer	Same as 7-1 Thor/Delta				
7-2	Tanks	Same as 7-2 Thor/Delta				
7-3	Fill and Drain Valve	Same as 7-3 Thor/Delta				
7-4	Thrusters	Same as 7-4 Thor/Delta				
7-5	Connectors and Manifold	Same as 7-5 Thor/Delta				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, ATLAS/CENTAUR SUBSYSTEM STRUCTURE/THERMAL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
8-1	Structure	Same as 8-1 Thor/Delta				
8-2	Louvers	Same as 8-2 Thor/delta				

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA SUBSYSTEM ELECTRICAL POWER

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
1-1	Solar Array	See 1-1 of Thor/Delt	Probe Bus			
1-2	Power Control Unit	See 1-2 of Thor/Delt	Probe Bus			
1-3	Shunt Radiator	See 1-3 of Thor/Delt	Probe Bus			
1-4	Battery	Provide power through mission for pulse loads and during eclipse	Loss of battery cell open or short	Loss of spacecraft power may loose mission due to inadequate thermal control or electronic box scramble when power comes back up due to array	Telemetry	Strict quality control on battery
1-5	DC-DC Converter	See 1-5 of Thor/Delt	Probe Bus			

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA

SUBSYSTEM COMMUNICATIONS

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
2-1	Forward Omni	See 2-1 of Thor/Delta	Probe Bus			
2-2	Aft Omni	See 2-2 of Thor/Delta	Probe Bus			
2-3	Diplexers	See 2-3 of Thor/Delta	Probe Bus			
2-4	Transfer Switch	See 2-8 of Thor/Delta	Probe Bus			
2-5	Receiver	See 2-5 of Thor/Delta	Probe Bus			
2-6	Medium Gain Antenna	See 2-6 of Thor/Delta	Probe Bus			
2-7	Transmitter Amplifier	See 2-7 of Thor/Delta	Probe Bus			
2-8	Transfer Switch	See 2-8 of Thor/Delta	Probe Bus			
2-9	Hybrid	See 2-9 of Thor/Delta	Probe Bus			
2-10	Transmitter	See 2-10 of Thor/Delta	Probe Bus			

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM PROBE BUS, THOR/DELTA SUBSYSTEM COMMUNICATIONS

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
2-11	High Gain Antenna	Provide downlink for science data and uplink	Loss of antenna	Loss of mission data	Telemetry	Structure item built with safety factor
2-12	Conscan	Provide antenna boresight	Loss of conscan	Loss of onboard conscan	Telemetry	Perform conscan open loop

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA SUBSYSTEM DATA HANDLING

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
3-1	Digital Decoding Unit	See 3-1 of Thor/Delt	Probe Bus			
3-2	Digital Telemetry Unit	See 3-2 of Thor/Delt	Probe Bus			
3-3	Digital Storage Unit	Provide data storage during occultation	Loss of a DSU	Loss of redundancy	Telemetry	Redundant DSU's 2 of 3 required

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA

SUBSYSTEM ELECTRICAL DISTRIBUTION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
4-1	Command Distribution Unit	See 4-1 of Thor/Delt	Probe Bus			

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA

SUBSYSTEM ATTITUDE CONTROL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
	See Thor/Delta Probe Bus					

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA SUBSYSTEM PROPULSION

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
7-1 to 7-5	See Thor/Delta	Probe Bus				
7-6	Insertion motor	Provide orbiter insertion	Fail to fire	Loss of mission	Telemetry	Stringent quality control

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, THOR/DELTA

SUBSYSTEM STRUCTURE/THERMAL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
8-1 to 8-2	See Thor/Delta Probe Bus					

Table 6I-4. System Failure Mode and Effects Analysis (Continued)

SYSTEM ORBITER, ATLAS/CENTAUR SUBSYSTEM ALL

FMEA ITEM 1	ITEM NOMENCLATURE 2	FUNCTION 3	FAILURE MODE/MECHANISM 4	FAILURE EFFECT ON SUBSYSTEM 5	METHOD OF DETECTION 6	CONTROLS IN EFFECT TO ELIMINATE OR REDUCE FAILURE MODE OCCURRENCE OR EFFECTS 7
	See Thor/Delta	Orbiter and modify	with changes in Atlas/	Centaur Probe Bus		